

R. Harcey

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AEROSPACE GROUP  
SPACE SYSTEMS DIVISION  
HUGHES AIRCRAFT COMPANY  
CULVER CITY, CALIFORNIA

HUGHES

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18 June 1963

SUBJECT: Advanced Syncom Monthly Progress  
Report for May 1963

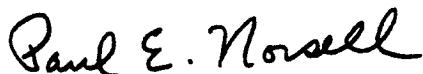
TO: Mr. Robert J. Darcey  
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Attached are copies of the Advanced Syncom Monthly Progress Report for May 1963.

Vibration tests of the engineering model structure were conducted with qualification level, sinusoidal inputs supplied at the spacecraft to Agena, and the spacecraft to apogee injection motor interfaces. The structure demonstrated design stiffness near the predicted values. The next brief series of tests on the structure will consist of exploratory studies to assess effects of locating access ports in the thrust tube and removal of various structural members. Subsequently, a modal survey and a torsional vibration test, with input supplied at the Agena interface, will be conducted.

The Marquardt Corporation conducted a formal program status briefing and feasibility demonstrations of a developmental Syncom II reaction control system for NASA Goddard Space Flight Center and Hughes representatives on 24 March 1963. The demonstrations were effective and the status report indicated that significant progress has been made by Marquardt in resolving technical design and test instrumentation problems encountered early in the development program.

HUGHES AIRCRAFT COMPANY



Paul E. Norsell  
Manager, Systems Development  
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# *Advanced SYNCOM*

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*May 1963*

## *MONTHLY PROGRESS REPORT*

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*NASA Contract 5-2797  
SSD 3339R*

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## 1. INTRODUCTION

The use of communication satellites has been recognized to answer the need for greatly expanded global communications capability. It has been a major effort of the United States Government and of industry to develop a satellite relay system at the earliest possible time.

Under NASA Goddard Space Flight Center Contract NAS-5-1560, Hughes Aircraft Company developed the Syncom I spacecraft to be orbited by NASA Delta launch vehicles and used in conjunction with Department of Defense Advent ground stations for the performance of inclined synchronous-orbit communication experiments during 1963.

The Syncom I spacecraft will demonstrate a simple spin-stabilized design capable of being placed in a synchronous orbit. At the same time, it will be demonstrated that a simple pulse-jet control system can provide the stationkeeping necessary to maintain a synchronous orbit.

Additional important mission objectives of the NASA communication satellite program include the demonstration of a "stationary" or equatorial, synchronous orbit, conduct of system orbital life tests, demonstration of new wide-band services on a transoceanic basis, and demonstration of a system accessible to all nations.

Under NASA Goddard Space Flight Center Contract NAS-5-2797, Hughes is conducting feasibility studies and advanced technological development for an advanced, stationary, active repeater communication satellite. A Summary Report covered the technical progress achieved during the original contract period and details the system configuration resulting from the system studies. A subsequent supplementary report covered further studies made under modification two to the above contract and the accompanying technical direction.

This report covers contractual activities during the month of May, 1963.

## 2. COMMUNICATION SUBSYSTEM

Determination was made of the ratio of carrier power to total power as a function of rms modulation index for a carrier phase modulated by random noise.

In the multiple-access mode of the spacecraft-to-ground link the carrier is phase modulated by a signal with the characteristics of random noise. To determine the feasibility of tracking the carrier using a phase-locked tracking receiver the amount of power in the carrier as a function of the rms modulation index should be determined.

Assume a carrier is phase modulated by random noise of bandwidth B. The rms modulation index is therefore the rms value of the phase deviation of the carrier:

$$\text{rms modulation index} = \bar{\theta}$$

$$\text{or peak modulation index } M = \sqrt{2} \bar{\theta}$$

The mean square value of the modulation index is proportional to signal power, which in this case is proportional to the bandwidth B of the modulating spectrum (noise).

$$\bar{\theta}^2 = KB$$

It may be assumed that B is the sum of N contiguous narrow bands of noise, each of width B/N. Therefore each of the N bands contributes  $\bar{\theta}^2/N$  to the mean square value of the modulation index, or the rms modulation index due to a narrow band is  $\sqrt{\bar{\theta}^2/N}$ . Now the power is represented in each small noise band by a sine wave of the same equivalent power of peak amplitude  $M_n$ . The peak modulation index is thus

$$M_n = \sqrt{2} \sqrt{\frac{\bar{\theta}^2}{N}} = \bar{\theta} \sqrt{\frac{2}{N}}$$

The ratio of carrier power to total power in a signal modulated by N sinusoidal subcarriers is

$$\frac{P_c}{P_{tot}} = \prod_{n=1}^N \left[ J_0(M_n) \right]^2$$

If all N signals have the same modulation index M,

$$\frac{P_c}{P_{tot}} = \left[ J_0(M) \right]^{2N}$$

For small values of M,  $J_0(M)$  may be approximated by

$$J_0(M) \approx 1 - \frac{M^2}{4}$$

Therefore

$$\frac{P_c}{P_{tot}} \approx \left( 1 - \frac{M^2}{4} \right)^{2N} = \left( 1 - \frac{2\bar{\theta}^2}{4N} \right)^{2N} = \left( 1 - \frac{\bar{\theta}^2}{2N} \right)^{2N}$$

To simplify this, consider the expression

$$\begin{aligned} \ln \left( 1 - \frac{\bar{\theta}^2}{2N} \right)^{2N} &= 2N \ln \left( 1 - \frac{\bar{\theta}^2}{2N} \right) \approx 2N \left( -\frac{\bar{\theta}^2}{2N} \right) \\ &= -\bar{\theta}^2 \quad \left( \text{for } \frac{\bar{\theta}^2}{2N} \text{ small} \right) \end{aligned}$$

Therefore

$$\left( 1 - \frac{\bar{\theta}^2}{2N} \right)^{2N} \approx e^{-\bar{\theta}^2}$$

or

$$\frac{P_c}{P_{tot}} \approx e^{-\bar{\theta}^2}$$

or

$$10 \log \frac{P_c}{P_{\text{tot}}} = -4.34 \bar{\theta}^2 \text{ decibels}$$

This relationship is plotted in Figure 2-1.

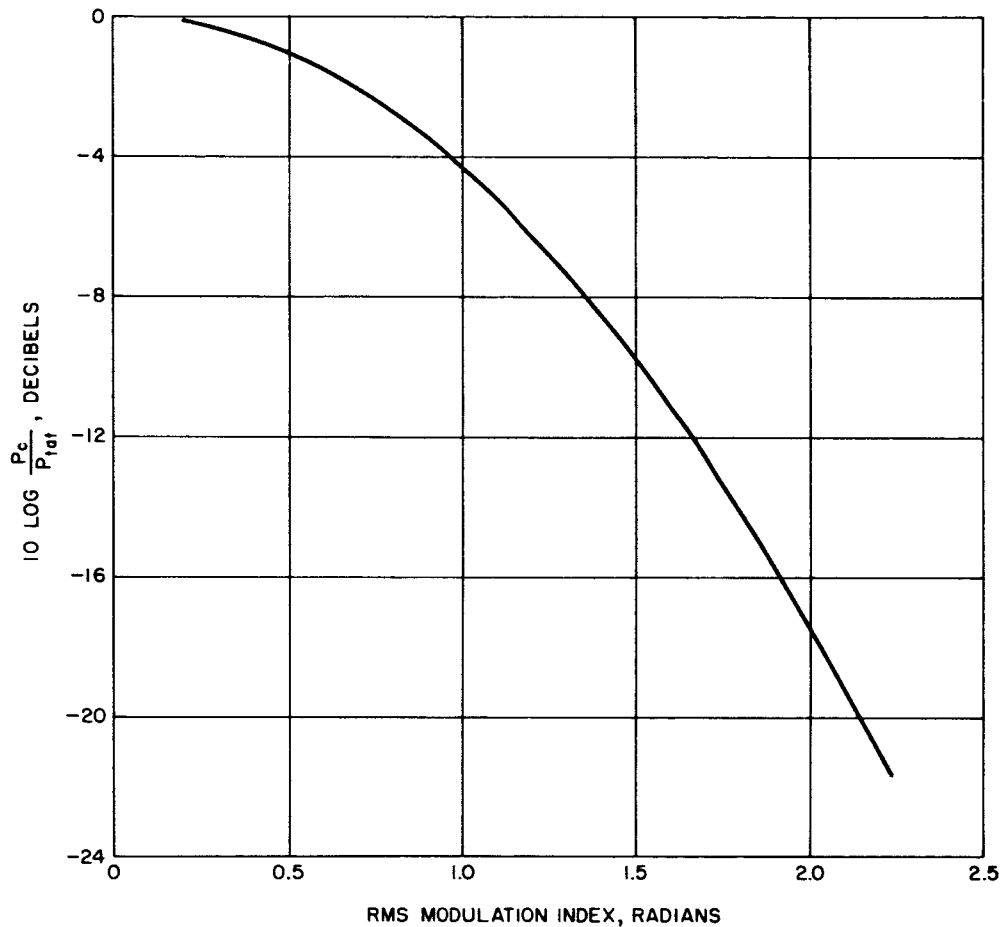


Figure 2-1. Ratio of Carrier Power to Total Power as Function of RMS Modulation Index for Carrier Phase Modulated by Random Noise

### 3. LAUNCH AND ORBIT ANALYSIS

#### CONSIDERATIONS OF FIXED VERSUS VARIABLE APOGEE IGNITION TIMER

This section will assist in ascertaining the errors in the synchronous orbit resulting from planned changes of parking orbit altitude without a corresponding correction in the time from Agena II separation to apogee motor firing. The noncorrectable timing device employed for apogee motor firing is referred to as a fixed timer.

The use of such a fixed timer in conjunction with any planned parking orbit altitude change will necessarily result in apogee motor firing at a time other than apogee; i.e., a timer error exists. The purpose of this section, therefore, is to decide if the magnitude of the resulting errors are significant to the extent that a variable timer is required.

To obtain the magnitude of the resulting errors, a Syncom I Keplerian apogee firing computer program (FUSIT) (Reference 3-1) was modified by Syncom II apogee firing geometry (Figure 3-1) and run for a timer error of  $T = \pm 0, 1, 2, \dots, 10$  minutes. Other parameters used for the calculations are shown in Table 3-1. Residual errors are present at  $T = 0$  due to inaccuracies in nominal parameters.

The results of immediate interest are shown in Table 3-2. An examination of the plot of  $|V_d|$  (velocity correction necessary to remove drift) versus  $T$  (time from apogee), Figure 3-2, shows that  $|V_d| \cong kT^2$  where  $k \cong 0.02 \text{ fps/min}^2$ .

The  $3\sigma$  ascent guidance errors, including an inherent 2-minute timer error, require a drift velocity correction,  $|V_d|$ , of 120 fps (Reference 3-2). The tabulated data in Table 3-2 shows that, for a  $\pm 10$ -minute timer error, an additional  $|V_d|$  of 1.79 fps is required. This is an increase of approximately 1.49 percent. Therefore, the additional timer error causes an insignificant increase in the required drift velocity correction.

An examination of the plot of the velocity correction (Figure 3-3) necessary to remove inclination,  $V_i$ , shows the  $\Delta V_i / \Delta T \cong 11.1 \text{ fps/min}$ .

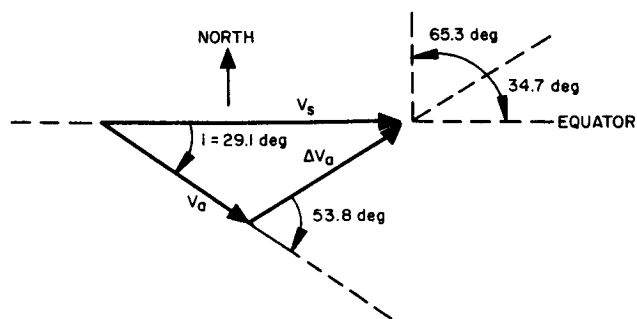


Figure 3-1. Apogee Motor Boost Geometry

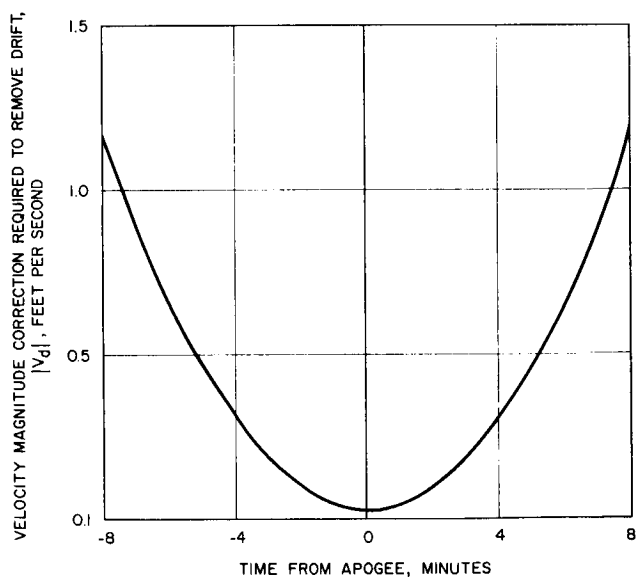


Figure 3-2. Velocity Magnitude Correction Required to Remove Drift,  $|v_d|$ ,  $v_s$ , Time from Apogee,  $T$

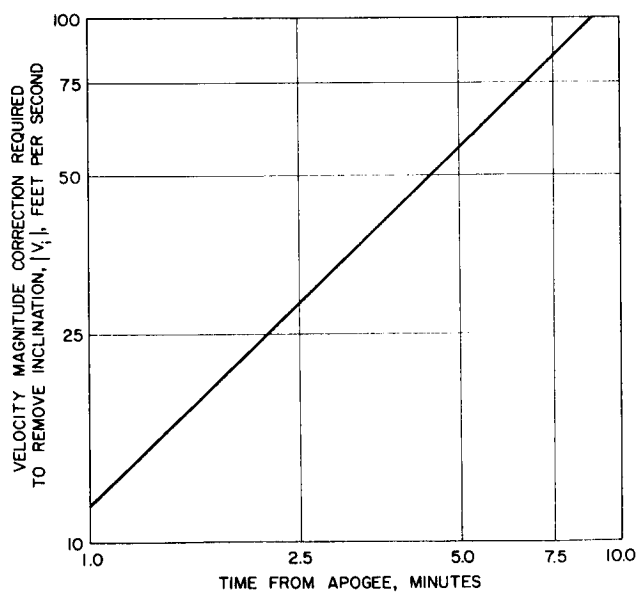


Figure 3-3. Velocity Magnitude Correction Required to Remove Inclination,  $v_i$ ,  $v_s$

TABLE 3-1. ORBITAL PARAMETERS

T at apogee = 0

Transfer Ellipse

|                             |                                    |
|-----------------------------|------------------------------------|
| Apogee radius               | 22, 752.3 n.mi.                    |
| Apogee longitude            | 93° W (descending node)            |
| Apogee velocity, $V_a$      | 5250 fps (0.863 n.mi./sec)         |
| Perigee radius              | ~ 3529.7 n.mi. (88 n.mi. altitude) |
| Perigee velocity            | ~33, 760 fps                       |
| Semimajor axis              | 13.141 n.mi.                       |
| Nominal period, $\tau$      | 37, 800 seconds                    |
| Inclination                 | 29.1 degrees                       |
| Longitude of ascending node | ~ 166.5 degrees E                  |
| Period                      | ~ 37, 800 seconds (~10.51 hours)   |

Apogee Boost,  $\Delta V_a$  (descending node)

|                              |                              |
|------------------------------|------------------------------|
| $\Delta V_a$                 | 6075 fps                     |
| Burning time                 | ≈ 45 seconds                 |
| $V_s$ , synchronous velocity | ≡ 10,087.5 fps               |
| Elevation                    | ~ 0 degrees                  |
| Azimuth                      | 65.3 degrees from North (CW) |

Timer Setting  $\frac{T}{2} \pm N$  minute,  $N = 0, 1, 2, \dots, 10$



A  $\pm 10$ -minute timer error requires an additional  $|V_i|$  of 111 fps. This correction is clearly of significant value.

The significance of the additional velocity correction required must be assessed with respect to the change in planned parking orbit altitude necessary to cause such a timer error. The correlation of planned parking orbit altitude change and resulting timer error is shown in Table 3-2.

TABLE 3-2. EFFECTS OF TIMER SETTING ON PERIGEE CHANGE

|  |   |
|--|---|
| $\Delta T$   | Change in one-half period (timer error) |
| $T = 315$ minutes  | One-half nominal period                 |
| $\Delta r_p$   | Change in parking orbit altitude        |
| $a = 13,141$ n.mi.   | Nominal semimajor axis                  |
| $\Delta r_s$   | Change in synchronous radius            |
| $\Delta a$   | Change in semimajor axis                |
| $\frac{\Delta T}{T} = \frac{3}{2} \frac{\Delta a}{a}$                                |   |
| $\Delta a = \Delta r_p + \Delta r_s$   |   |
| $\Delta r_s \equiv 0$  |   |
| $\therefore \Delta r_p = \Delta a$   |   |
| $\Delta T/T = (3/2) (\Delta r_p/a)$  |   |
| $\Delta r_p = (\Delta T/T) \left(\frac{2}{3}\right) (a)$                             |   |
| $\Delta r_p = \left(\frac{2}{3}\right) \left(\frac{1}{315}\right) (13,141) \Delta T$ |   |
| $\Delta r_p = 27.8 \text{ n.mi./min}$  |   |

Table 3-3 shows a 10-minute timer error will result from a planned change in parking orbit altitude of 278.0 n.mi. This is clearly an unrealistic change of plan. A more realistic change in altitude would be at most 30 n.mi.

TABLE 3-3. TIMER ERROR EFFECTS

| T<br>From Apogee,<br>minutes | $ \Delta r_p $ , n.mi. | $ V_i $ , fps | $ V_d $ , fps | Eccentricity |
|------------------------------|------------------------|---------------|---------------|--------------|
| 10                           | 278.0                  | 111           | 1.79          | 0.023        |
| 9                            | 250.2                  | 100           | 1.46          | 0.021        |
| 8                            | 222.4                  | 89.0          | 1.16          | 0.019        |
| 7                            | 194.6                  | 77.9          | 0.842         | 0.016        |
| 6                            | 166.8                  | 66.8          | 0.66          | 0.014        |
| 5                            | 139.0                  | 55.7          | 0.465         | 0.011        |
| 4                            | 111.2                  | 44.7          | 0.306         | 0.0095       |
| 3                            | 83.4                   | 33.6          | 0.186         | 0.0071       |
| 2                            | 55.6                   | 22.5          | 0.093         | 0.0047       |
| 1                            | 27.8                   | 11.7          | 0.046         | 0.0023       |
| 0                            | 0                      | 4.0           | 0.0279        | 0.00001      |
| - 1                          | 27.8                   | 11.7          | 0.046         | 0.0023       |
| - 2                          | 55.6                   | 22.5          | 0.093         | 0.0047       |
| - 3                          | 83.4                   | 33.6          | 0.186         | 0.0071       |
| - 4                          | 111.2                  | 44.7          | 0.306         | 0.0095       |
| - 5                          | 139.0                  | 55.7          | 0.465         | 0.011        |
| - 6                          | 166.8                  | 66.8          | 0.66          | 0.014        |
| - 7                          | 194.6                  | 77.9          | 0.842         | 0.016        |
| - 8                          | 222.4                  | 89.0          | 1.16          | 0.019        |
| - 9                          | 250.2                  | 100           | 1.46          | 0.021        |
| -10                          | 278.0                  | 111           | 1.79          | 0.023        |

For ease of calculation, assume the change to be 27.8 n. mi. This corresponds to a 1-minute timer error. The resulting  $|V_d|$  is 0.046 fps and  $|V_i|$  is 11.7 fps.  $|V_d|$  is clearly insignificant.  $|V_i|$  is a 10.4 percent increase in the  $|V_i|$  of the  $3\sigma$  ascent guidance errors, which is not significant.

This study indicates that errors resulting from changes in planned parking orbit altitude with a fixed timer are not significant to the extent that a variable timer is required. If it should be desirable to reduce the slight change of  $|V_i|$  resulting from the use of a fixed timer, a change can be made in the Agena II yaw programmer, a readily variable unit.

## CONSIDERATIONS OF PULSED OPERATION OF AXIAL JET FOR INCLINATION REMOVAL

A possible excess temperature rise problem associated with the initial continuous operation of the axial hot gas jet has led to consideration of reducing the duty cycle of the axial jet during initial inclination removal sequence to ensure that maximum operating temperature is kept below an acceptable value.

The resultant increase in time for inclination removal is shown to have a negligible effect on the efficiency of inclination removal when compared with the continuous operation sequence.

From Reference 3-2 the  $3\sigma$  value of postapogee boost inclination is  $\Delta i \leq 0.65$  degree, so that the required velocity increment from the axial jet is

$$\Delta V_i \leq 176 \text{ fps/deg } (0.65 \text{ degree}) \approx 115 \text{ fps} \quad (3-1)$$

Now, from the relation of total impulse  $I_T$  with  $\Delta V_i$

$$I_T \cong F t_a = I_{sp} W_p = I_{sp} W_s \left[ 1 - \exp \left( - \frac{\Delta V_i}{g I_{sp}} \right) \right] \quad (3-2)$$

where

$F$  = average thrust of axial jet, pounds

= 5 pounds

$t_a$  = action time of axial jet for continuous operation

$I_{sp}$  = specific impulse of axial jet, second

= 290 seconds

$W_p$  = propellant weight, pounds

$W_s$  = spacecraft initial weight, pounds

$$\cong 765 \text{ pounds}$$

so that

$$W_p = 765 [1 - 0.98772] = 9.38 \text{ pounds}$$

$$I_T = F t_a = I_s W_p = (290) (9.38) = 2720 \text{ pounds-second}$$

and

$$t_a = \frac{2720}{5} = 544 \text{ seconds}$$

$$\cong 9.07 \text{ minutes}$$

If a reduced duty cycle is required, it is desirable to pulse the axial jet "on" for an integer multiple of spin periods  $\tau_s$  ( $\tau_s \approx 0.6$  second) to minimize the magnitude of the induced precession angle  $P$  per on-off cycle. A tentative duty cycle of one-third, with the jet turned on for  $n$  spin periods and off for  $2n$  periods, will increase the time to remove the above inclination to

$$t_i \cong 3 t_a = 27.2 \text{ minutes} = 1632 \text{ seconds}$$

Thus, if 1/2 hour is allowed for inclination removal and this time interval is centered about the time at which the descending node of the orbit occurs (axial jet pointing south), the maximum latitude  $L$  encountered during this time is only

$$L \approx i \omega_e \frac{t_i}{2} \leq (11.4 \times 10^{-3}) (15 \text{ deg/hr}) (1/4 \text{ hour}) \cong 0.043 \text{ degree}$$

This value is negligible compared with the uncertainty of the spin axis attitude ( $\sim 1$  degree) so that the impulsive assumption used for  $\Delta V_i$  is still valid since  $t_i \ll 24$  hours.

Further investigations will be made of the magnitude of the induced nutation angle  $\theta$  per on-off cycle. An additional residual torque pulse per on-off cycle may accrue as a result of mismatching the rise and decay thrust transients separated in time by  $n \tau_s$  ( $n = 1, 2, \dots$ ), where  $\tau_s$  is the spin period. For a thrust rise transient ( $\sim 12$  millisecond) ramp that is almost equal to the negative of the decay ramp, the residual torque pulse can be made small by adjusting the jet on time to an exact integer multiple of  $\tau_s$ . If the rise transient (plus dead time delays) is not exactly the mirror image of the decay transient, a calibration procedure may be desirable to ascertain

the relative spin angle  $\psi$  between the on and off commands that will result in the smallest torque pulse per on-off cycle. In addition, since the number of on-off cycles required for one-third duty cycle may be estimated by

$$\text{Number of on-off cycles} \approx \frac{t_i}{3 n \tau_s} \approx \frac{908}{n} \quad ; \quad n = 1, 2, \dots, N$$

it is desirable to make  $n$  as large as possible consistent with an upper safe temperature-bound corresponding to some integer  $N$ .

#### REFERENCES

- 3-1. "Syncom I 7090 Computer System Description - FUSIT," Hughes IDC 2280.03/49, 5 September 1962.
- 3-2. "Syncom II Summary Report," Hughes SSD 3128R, NASA Contract 5-2797, 31 March 1963.

#### 4. SPACECRAFT SYSTEMS DESIGN

##### SPACECRAFT SUBSYSTEM PERFORMANCE REQUIREMENTS AND BLOCK DIAGRAMS

The following revisions and changes were made to update the Subsystem Functional Performance Specification, dated April 1963. In addition, functional diagrams have been revised and are shown in Figures 4-1, 4-2, and 4-3.

3.2.2.4.3 Power out - Each receiver shall have a power out of 1.1 mw  $\pm$  mw.

Replace Table 6.2.

Replace paragraphs 3.2.2.5.2 and 3.2.2.5.3.

3.2.2.5.2 Received Carrier/Received Noise - The requirements of this specification shall not be imposed unless the received carrier power to the received noise power ratio is at least +20.1 dbw.

3.2.2.5.3 Frequency Stability - The beacon signal frequency shall be stable to within 0.002 percent.

Replace paragraphs 3.2.4.2.1 and 3.2.4.2.2.

3.2.4.2.1 Losses - The losses including the power split shall not exceed 3.40 db over the frequency range 3992.09 to 4194.49 mc.

3.2.4.2.2 RF Power Input - The RF power input shall be 1.1  $\pm$  milliwatt.

Add to paragraph 3.2.4:

3.2.4.3.6 Filter - An S-band output filter shall be provided.

Replace paragraph 3.2.4.6.1.

3.2.4.6.1 Quantity - There shall be two RF switches, 475173.

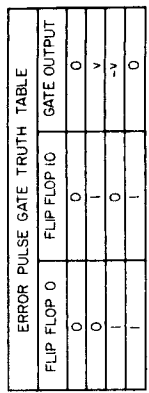
TABLE 6-2. TRANSPONDER - 475025  
Inputs and Outputs

| Control Item | Number of<br>Quadrants | Function  | Description   |
|--------------|------------------------|---|---------------|
| Inputs from: |                        |   |               |
| 475173       | 1                      | Multiple access                                       | AM/SSB        |
| 475173       | 1                      | Frequency translation                                 | WBFM          |
| 475211       | 4                      | Turn on multiple-access<br>series regulator           | Command pulse |
| 475211       | 4                      | Turn off multiple-access<br>series regulator          | Command pulse |
| 475211       | 4                      | Turn on frequency<br>translation series<br>regulator  | Command pulse |
| 475211       | 4                      | Turn off frequency<br>translation series<br>regulator | Command pulse |
| Outputs to:  |                        |   |               |
| 475171       | 1                      | Multiple-access output                                | PM            |
| 475171       | 1                      | Frequency translation<br>output                       | WBFM          |
| 475221       | 4                      | Multiple-access signal<br>strength                    |               |
| 475221       | 4                      | Frequency translation<br>signal strength              |               |

The standard command pulse shall be 60 msec long, have a 5-msec rise time, and be 0.2 volt in amplitude.

Replace paragraphs 3.2.4.6.2 through 3.2.4.6.6.

3.2.4.6.2 RF Switch No. 1



1-3



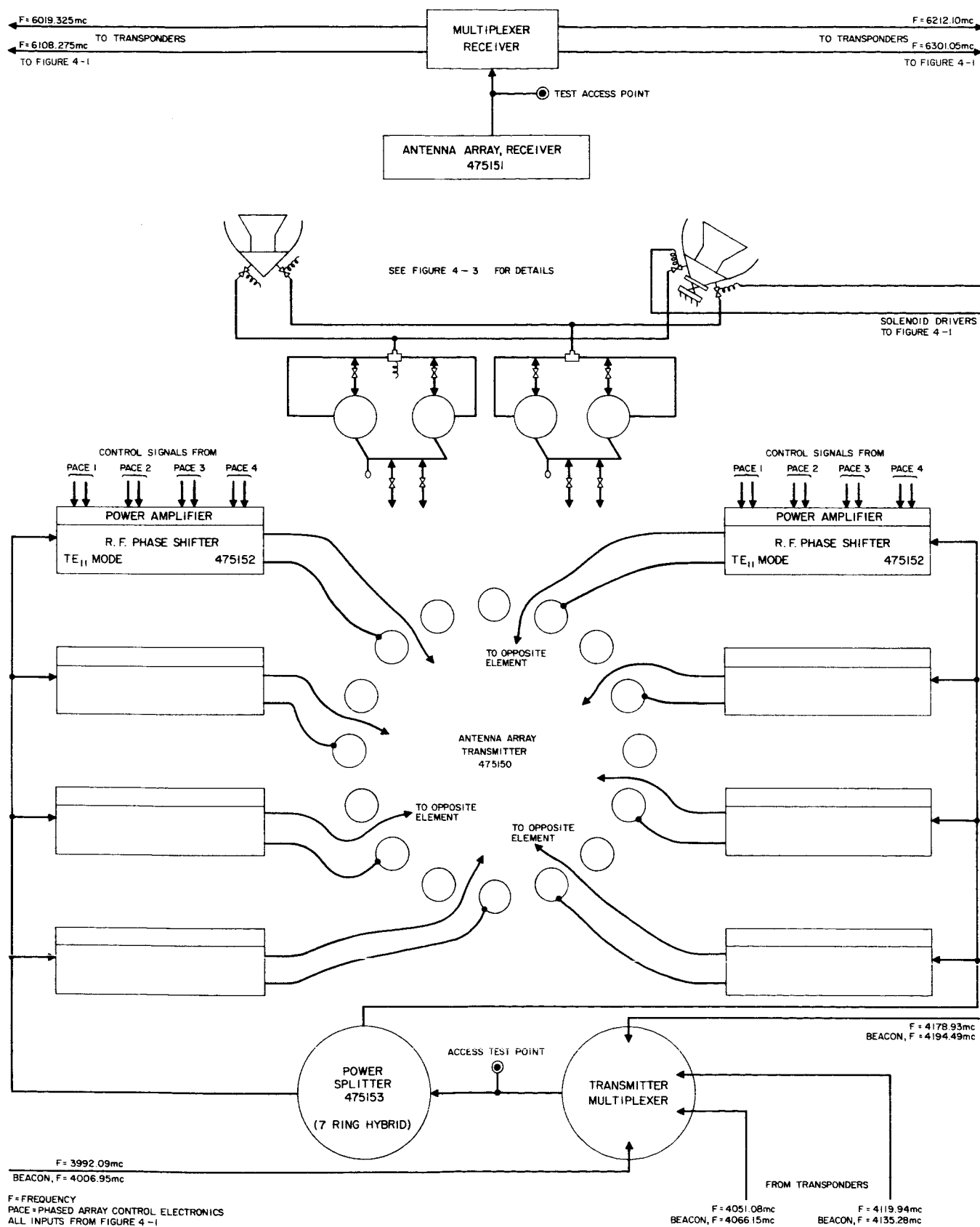


Figure 4-2. Advanced Syncom Block Diagram No. 2

3.2.4.6.2.1 RF Power In - The RF power in shall be at least -100.2 dbw.

3.2.4.6.2.2 Frequency Band - The RF switch No. 1 shall be capable of operating within specifications over the frequency range of 6019.325 to 6301.05 mc.

3.2.4.6.2.3 Losses - The losses shall not exceed 0.3 db over the frequency range given in 3.2.4.6.2.2.

3.2.4.6.2.4 Switching Current - The switching current shall not exceed 1.0 ampere.

3.2.4.6.2.5 Input and Output Impedance - The input and output impedance shall be made as closely as possible to 50 ohms.

Add to paragraph 3.2.4.6:

3.2.4.6.3 RF Switch No. 2.

3.2.4.6.3.1 RF Power Input - The RF power input shall be at least 4 watts.

3.2.4.6.3.2 Frequency Band - The RF switch shall be capable of operating within specification over the frequency range, 3992.09 to 4194.49 mc.

3.2.4.6.3.3. Losses - The losses shall not exceed 0.3 db over the frequency range 3992.09 to 4194.49 mc.

3.2.4.6.3.4 Switch Current - The switching current shall not exceed 1.0 ampere.

3.2.4.6.3.5 Input and Output Impedance - The input and output impedance shall be made as close as possible to 50 ohms.

3.2.4.6.4 Input-Outputs - RF switch input-outputs shall be as given in Table 6-7.

Replace Table 6-7.

Add to paragraph 3.3.3.2:

3.3.3.2.1 Output Level - The output level shall be -24 volts.

Add to paragraph 3.3.3.3:

3.3.3.3.1 Output Level - The output level shall be -24 volts.

3.3.3.3.3.2 Pulse Width - The pulse shall be 10 microseconds wide.

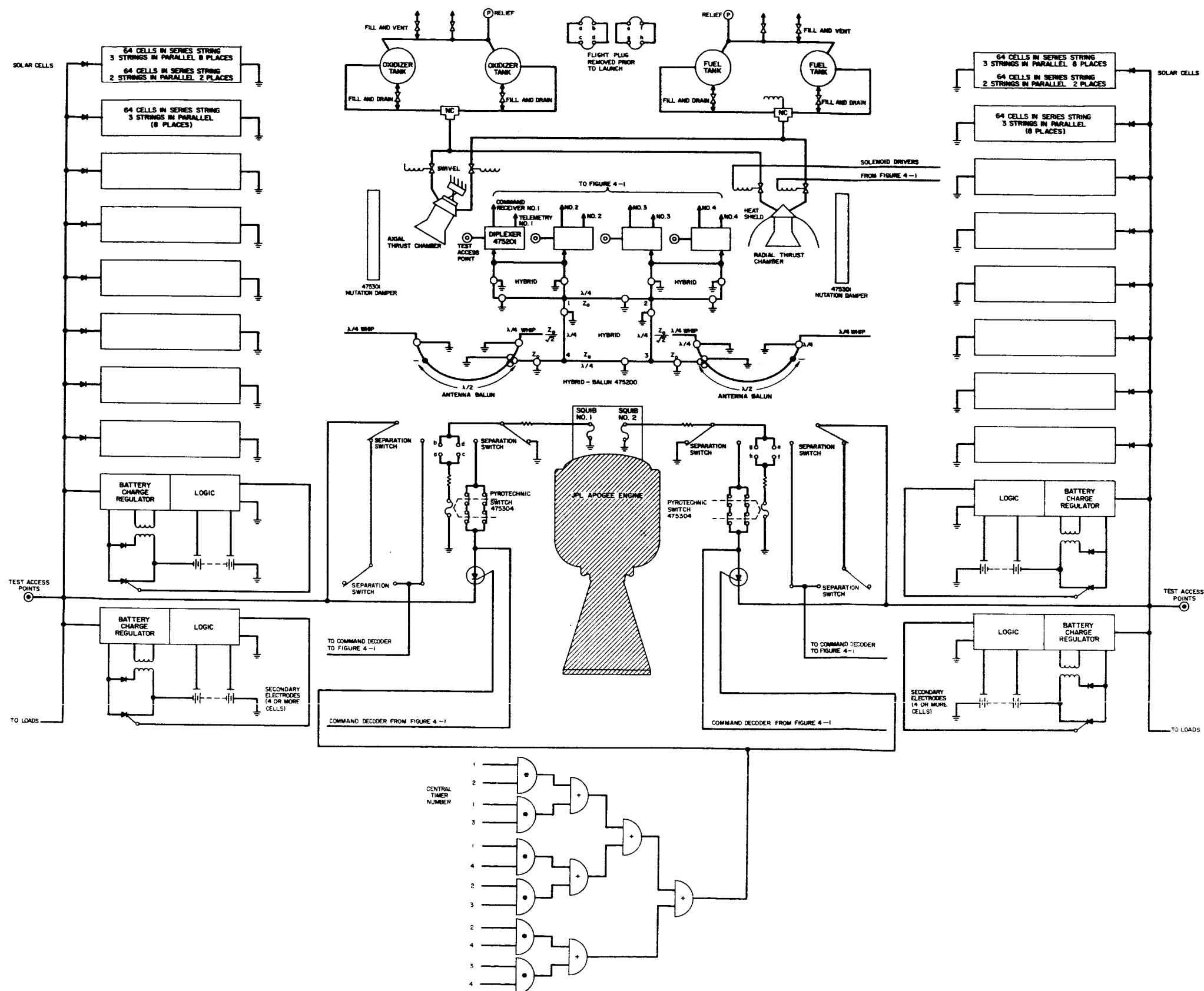


Figure 4-3. Advanced Syncom Subsystem Block Diagram No. 3

TABLE 6-7. RF SWITCH - 475173  
Inputs and Outputs

| Control Item        | Number of<br>Quadrants | Function             | Description                |
|---------------------|------------------------|----------------------|----------------------------|
| <u>Switch No. 1</u> |                        |                      |                            |
| Inputs from:        |                        |                      |                            |
| Multiplexer         |                        | RF<br>Switch current | 6 gc -1002 dbw<br>1 ampere |
| Outputs to:         |                        |                      |                            |
| 475025              | 1                      | RF multiple access   | 6019 - 6301 mc             |
| 475025              | 1                      | RF translation       | 6010 - 6301 mc             |
| <u>Switch No. 2</u> |                        |                      |                            |
| Inputs from:        |                        |                      |                            |
| TWT 384H No. 1      | 1                      | RF                   | 4 gc 6 dbw                 |
| TWT 384H No. 2      | 1                      | RF                   | 4 gc 6 dbw                 |
| 475174 No. 1        | 1                      | Switch               | 1 ampere                   |
| 475174 No. 2        | 1                      | Switch               | 1 ampere                   |
| Outputs to:         |                        |                      |                            |
| 4715154             | 1                      | RF                   | 5.7 dbw                    |

Add to Table 6-8:

| Control Item | Number of<br>Quadrants | Function | Description   |
|--------------|------------------------|----------|---------------|
| Input from:  |                        |          |               |
| 475211       | 4                      | Pseudo 4 | Command pulse |

Add to Table 6-10:

| Control Item         | Number of Quadrants | Function | Description   |
|----------------------|---------------------|----------|---------------|
| Output to:<br>475160 | 4                   | Psuedo 4 | Command pulse |

Replace paragraphs 3.5.2 through 3.5.2.4.1.

3.5.2 Battery Regulators

3.5.2.1 Quantity - There shall be four battery regulators.

3.5.2.2 Impedance - The battery regulator shall be compatible with the solar array impedance characteristics and those of the unregulated bus.

3.5.2.3 Switching - The battery regulator shall provide necessary switching of the battery to the unregulated bus to permit satisfactory spacecraft operation under eclipse and transient load periods.

3.5.2.4 Current Limiting - The battery regulator shall provide current limiting to the battery charging network to eliminate any excessive unregulated bus drain in the event of a battery failure.

3.5.2.5 Current Availability - The battery regulator shall enable the solar array to charge the battery regardless of maximum normal electronics operation.

3.5.2.6 Sensory Electrode Operation - The battery regulator shall provide satisfactory operation with or without battery cells containing a sensory electrode.

Add to paragraph 3.5.4:

3.5.4.7 High Current Discharge - The cell terminal voltage during discharge of a fully-charged cell at a load of 12.0 amperes shall be 1.0 volt minimum for a period of 10 seconds.

Add to paragraph 3.5.4:

3.5.4.8 Capacity at High Rate Discharge - The cell discharge capacity at 75° F shall be a minimum of 4.8 ampere-hours when discharged at a constant current of 6.0 amperes to an end voltage of 1.0 volt.

Replace paragraph 3.7.2.

3.7.2 Harness Construction - The harness shall be constructed to minimize noise effects and to minimize physical damage from heat, flexing, and centrifugal force.

## COMMUNICATION TRANSPONDERS

### General Discussion

During this report period, the major change in the transponder system block diagram consisted of redesign of the multiple-access master oscillator to meet 8 cps/sec maximum drift rate during the eclipse period. Also, gain changes for engineering convenience were made in the preamplifier and filter amplifier. (See Transponder System Block Diagram.)

### Transponder Components Status

#### Circuits Common to Frequency Translation and Multiple-Access Transponders

Ferrite Switch, 6 gc. The breadboard design of this unit is being fabricated.

Mixer, 6 gc. Eight operational configurations of this unit are scheduled for fabrication. At present the ground planes and etched circuits for all these units are undergoing fabrication.

Bandpass Filter, 6 gc. The first modified unit (Figure 4-4) is undergoing test. The diameter was increased to lower the resonant frequency of the unwanted TM 010 mode from 6.0 to 5.230 gc and thus increase mode separation.

X3 Multiplier. This unit is undergoing slight design modification to facilitate fabrication. Manufacturing drawings are being made.

X32 Multiplier. This unit is now being fabricated. Retrofit changes are being designed for the last two doubler stages. The purpose of these changes is to ensure a tuning capability over the required range for all purposes with one design. Details of changes on the next to last doubler have been completed, and changes are partially completed in the last stage.

Isolators. Drawings are being prepared for the operational hardware. A magnetic shield is being added to allow close proximity of these units.

X2 Multiplier. The design of these units has been completed and fabrication initiated for the operational configuration. See Figure 4-5.

Hybrid, 3db. Drawings are being prepared. The external configuration is being modified to reduce size and weight.

Ferrite Switch, 4 gc. Redesign for the final configuration is being made. Drawings are being prepared.

Dual Coupler Detector. Coupler ground planes and circuit cards are being fabricated. Detector mounts have been ordered.

Bandpass Filter, 4 gc. Filters have been ordered from Rantec to meet the following specifications:

|                       |                          |
|-----------------------|--------------------------|
| Center frequency:     | a) 4051 mc<br>b) 4120 mc |
| Insertion loss:       | 1.0 db maximum           |
| VSWR over passband:   | 1.2 maximum              |
| Bandwidth (1.2 VSWR): | 25 mc minimum            |
| Rejection:            | a) 4114 mc<br>b) 4186 mc |
|                       | } 30 db minimum          |
| Weight:               | 3.0 ounces maximum       |
| Connector type:       | OSM                      |

Receiver Multiplexer. Two units have been ordered from Rantec to meet the following specifications:

|  |  |
|--|--|
| Center frequency:  | a) 6019 mc<br>b) 6108 mc<br>c) 6212 mc<br>d) 6301 mc |
| Insertion loss (all channels)<br>center frequency:               | 1.0 db maximum                                       |
| Insertion loss (all channels)<br>center frequency $\pm$ 12.5 mc: | 1.4 db maximum                                       |
| VSWR over passband:  | 1.30 maximum   |
| Rejection at 64 mc away<br>from center frequency:                | 22 db minimum  |
| Weight (tentative):  | 2.6 pounds   |
| Connector type:  | OSM  |





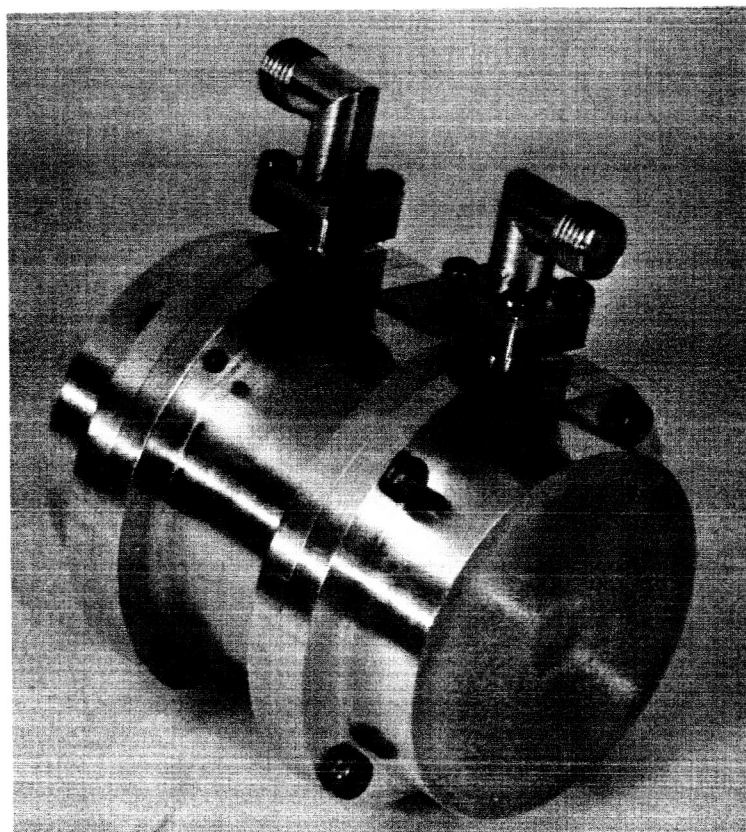


Figure 4-4. 69C Bandpass Filter

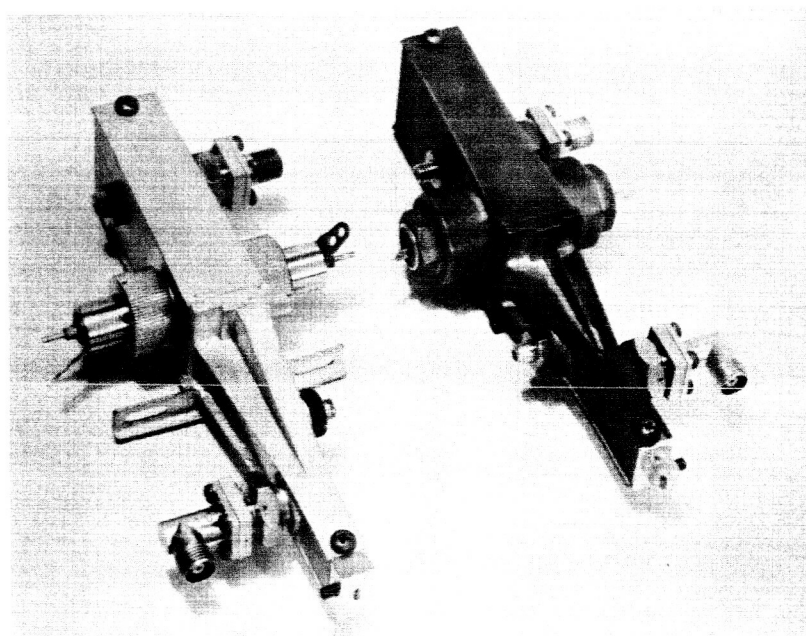


Figure 4-5. X2 Multiplier

Transmitter Multiplexer. Two units have been ordered from Rantec to meet the following specifications:

|   |  |
|---|--|
| Center frequency:                                     | a) 3992 mc<br>b) 4051 mc<br>c) 4120 mc<br>d) 4179 mc |
| Insertion loss (all channels)<br>over passband:       | 1.0 db maximum                                       |
| Insertion loss (all channels)<br>at beacon frequency: | 3.0 db maximum                                       |
| VSWR (passband):                                      | 1.30 maximum   |
| VSWR at beacon frequency:                             | 1.30 maximum   |
| Bandwidth (1.30 VSWR):                                | 25 mc minimum  |
| Rejection at $\pm 64$ mc from<br>center frequency:    | 30 db minimum  |
| Weight (tentative):                                   | 4.5 pounds   |
| Connector type:                                       | OSM  |

#### Circuits for the Multiple Access Transponder

Master Oscillator Amplifier. The master oscillator (Figure 4-6) is being repackaged in a separate thermally isolated box. This change is required to meet the 8 cps/sec drift requirement. The thermally isolated box will limit the temperature change to 5 °C/hour, which will meet the drift requirement even though the spacecraft may change 40 °C/hour during an eclipse. The master oscillator doubler, buffer, and amplifier stages will be packaged as a separate second unit.

Preamplifier. This unit is being modified to obtain a noise figure of 3.8 db over its passband.

Filter-Amplifier. The circuits have been completed and product engineering is under way (Figure 4-7).

Narrow Band Filter, 2 gc. Units for transponders have been ordered from Rantec to meet the following specifications:

|                   |                          |
|-------------------|--------------------------|
| Center frequency: | a) 2026 mc<br>b) 2060 mc |
|-------------------|--------------------------|

|                                       |                    |
|---------------------------------------|--------------------|
| Insertion loss over passband:         | 1.25 db maximum    |
| VSWR over passband:                   | 1.20 maximum       |
| Rejection 66 mc off center frequency: | 50 db minimum      |
| Bandwidth (1.2 VSWR):                 | 6 mc minimum       |
| Weight:                               | 2.5 ounces maximum |
| Connector type:                       | OSM                |

Wide Bandpass Filter, 2 gc. Units for transponders have been ordered from Rantec to meet the following specifications:

|  |                          |
|--|--------------------------|
| Center frequency:                      | a) 2026 mc<br>b) 2060 mc |
| Insertion loss over passband:          | 0.75 db maximum          |
| VSWR over passband:                    | 1.20 maximum             |
| Rejection 55 mc from center frequency: | 25 db minimum            |
| Bandwidth (1.2 VSWR):                  | 16 mc minimum            |
| Weight:                                | 2.5 ounces maximum       |
| Connector type:                        | OSM                      |

Phase Modulator. Circuit design has been completed and product design is being initiated.

Doubler Amplifier. Product design has been initiated following completion of the circuits.

#### Circuits for the Frequency Translation Transponder

Master Oscillator. Circuit design has been completed and product design has been initiated (Figure 4-8).

Preamplifier Intermediate-Amplifier, Postamplifier. These units are now ready for testing, contingent on delivery of 0.1-1 mfd tuning capacitors.

Limiter Amplifier. Design has been completed (Figure 4-9). Determination of output requirements is awaiting completion of the high-level mixer.

High-Level Mixer. The breadboard of this unit is now being fabricated.

Dual Filter Hybrid. Units for transponders have been ordered from Rantec to meet the following specifications:

|   |                          |
|---|--------------------------|
| Center frequency:   | a) 2057 mc<br>b) 2093 mc |
| Insertion loss at center frequency:                                       | 1.25 db maximum          |
| VSWR at center frequency:   | 1.20 maximum             |
| Rejection 40 mc off center frequency:                                     | 40 db minimum            |
| Hybrid output ratio:  | 0.0 db $\pm$ 0.5 db      |
| Hybrid directivity:   | 20 db minimum            |
| Isolation between filter output terminals at 40 mc from center frequency: | 90 db minimum            |
| Weight:   | 8.5 ounces maximum       |
| Connector type:   | OSM                      |

#### Transponder Regulators

##### Multiple-Access Mode Regulator (101 Unit)

This regulator was originally intended to be a Syncom I-type series regulator. However, updated requirements necessitate some redesign effort. The nominal 6144 mc output of the X3 multiplier (116 unit) cannot have a frequency drift rate in excess of 8 cps/sec. This frequency and drift rate are harmonically related (192:1) to the 32 mc master oscillator, which has an input voltage sensitivity of approximately 20 cps/volt. If one-fourth of the X3 multiplier total allowable frequency drift rate or 2 cps/sec is allowed for input dc voltage regulation, then the master oscillator input cannot change more than approximately 0.5 mv/sec. The multiple-access mode regulator circuit is being examined carefully to determine the degree of redesign necessary to ensure that this voltage rate of change will not be exceeded.

It has been tentatively decided to improve the 101 unit temperature regulation. This will be accomplished by replacing the present zener

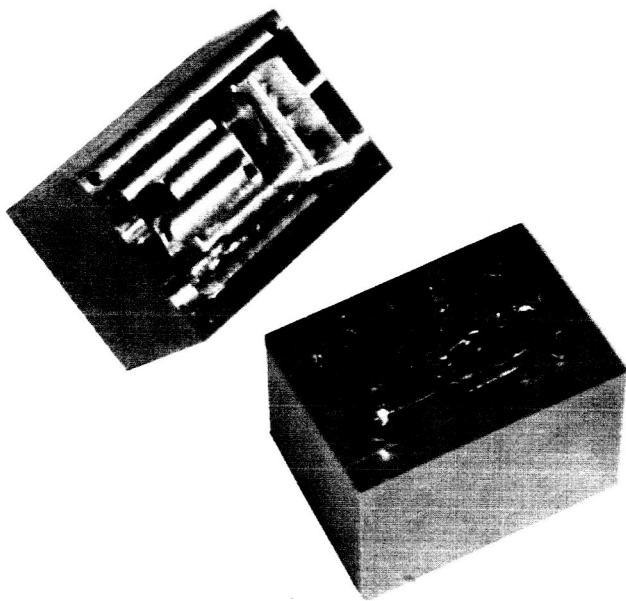


Figure 4-6. Master Oscillator  
Amplifier

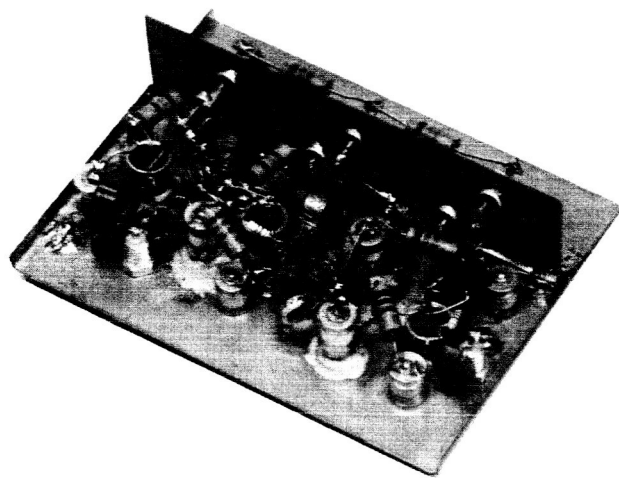


Figure 4-7. Filter Amplifier

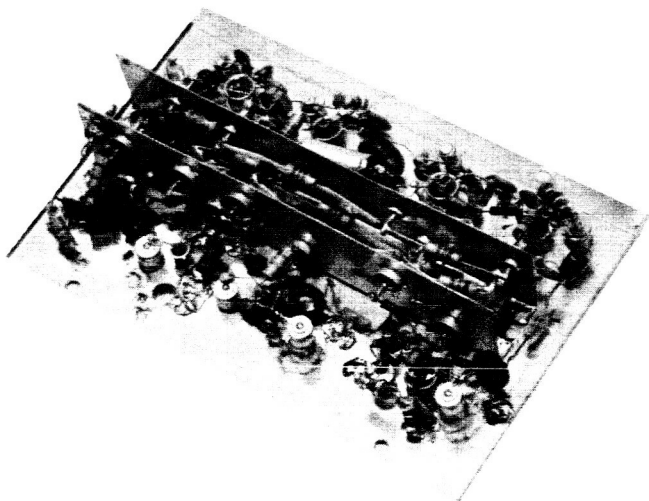


Figure 4-8. Master Oscillator

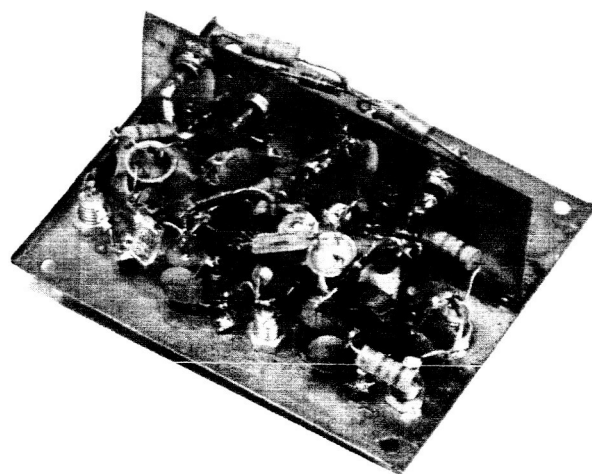


Figure 4-9. Limiter Amplifier

reference with a temperature compensated zener and by replacing the present single-ended first stage with a differential amplifier. This change should result in  $\pm 0.25$  percent static regulation for line changes between -26 and -36 volts, load changes between no load and full load (100 ma), and a 70° F temperature change. The dynamic regulation will be subsequently evaluated.

#### Frequency Translation Mode Regulator (102 Unit)

This unit will be a Syncom I-type series regulator, except for the addition of a sharp overcurrent trip circuit. A reasonable sharp overcurrent trip can be obtained with two resistors and a transistor. This circuit will use a resistive shunt to turn on the control transistor. The trip level is not as sharply defined as with the tunnel diode used in Syncom I command regulators, but it will be more than adequate.

It is necessary to add this simplified overcurrent trip to all series regulators to avoid excessive power dissipation in the series control element under overload conditions. The Syncom II unregulated bus has much more capacity than Syncom I. As such, the regulator input voltage can be maintained quite high under overload conditions; thus overload current must be provided.

### TRAVELING-WAVE TUBE POWER AMPLIFIER

#### Status

Major effort during this report period was divided between traveling-wave tube development and the tube life improvement program. Two tubes achieving 40 db saturated gain were constructed and tested. Several power supplies for the TWT life test rack were built and the power supplies and the life test rack for the diodes were designed.

The two tubes having increased gain were of the small helix diameter design and slightly longer than previous tubes. These tubes meet all of the specifications. One tube, No. 32, with the same pitch as 384H-27 operated at  $V_H = 1310$  volts and  $V_a = 125$  volts. While the efficiency of this tube was within the requirements, the margin of safety was not considered great enough for a production design. Curves of power out versus power in for constant beam voltage are shown in Figure 4-10. Tube No. 33 used a slightly smaller pitch (4 percent) and operated at  $V_H = 1200$  volts and  $V_a = 190$  volts. The efficiency was increased by this change.

Other TWT work included improvement of the RF match and redesign to accommodate OSM connectors. The RF match was greatly improved by changing the taper at the end of the helix. The standard OSM connectors were not mechanically adaptable to the TWT's pin match design; consequently, a modified version of the standard OSM connector is on order. Only minor adjustments remain to be made and the production of the required 32 tubes can be initiated.

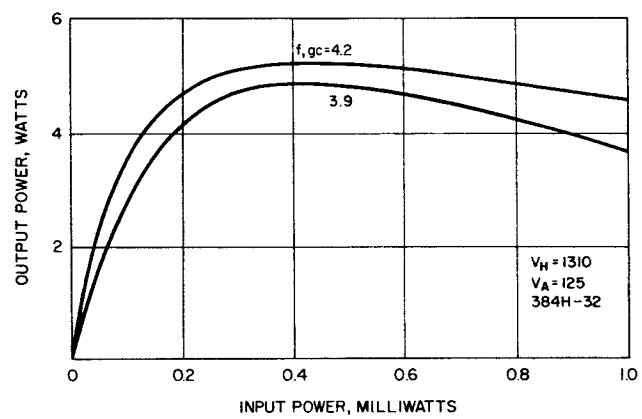


Figure 4-10. Constant Beam Voltage Curves

The parts required for the 32 tubes have been ordered and some parts have already arrived. In addition, about 50 cathode buttons have been ordered to begin chemical analysis of the effects of cleaning and processing on cathode impurities. Construction of the diodes for the life improvement program is temporarily delayed because of difficulties associated with the thermal couples for cathode temperature monitoring.

#### Qualification Test

The 7-day spin qualification test for the travelling-wave tube MTD 384H-No. 13 was completed and it met all test requirements. The tubes were oriented on the spin fixture in a position approximating the mounting angle on the spacecraft and were operated continuously on dc power. The accomplished test requirements were:

| <u>Acceleration</u> | <u>Rotational Speed</u> | <u>Time</u> |
|---------------------|-------------------------|-------------|
| 12 g                | 150 rpm                 | 7 days      |

The spin test completes the test program on tube MTD 384H-No. 13.

#### PHASED-ARRAY TRANSMITTING ANTENNA

Most of the drawings for the stripline version of the phase shifter circuits have been completed. A simplified cross-sectional drawing of the phase shifters is shown in Figure 4-11. The eight ferrite phase shifters are mounted vertically between the stripline circuits. The bottom circuit contains the eight-way power splitter and input couplers to the ferrite section, while the upper circuits include the output couplers from the ferrite section, the 90-degree hybrids, and the feed lines to the bases of the antennas. Altogether there are six ground planes which are die-stamped identically, although the hole patterns are different, and there are three different stripline circuit boards.

The die for stamping out the ground planes has been completed, and some samples have been made from aluminum plates, 0.020-inch thick. A stiffening bend is incorporated around the edge and in the center of the plates. The planes are stamped out while in a soft condition, and are then heat-treated to bring them to the proper stiffness. The first sample was satisfactory.

The art boards for the output coupler striplines were completed, the photographically reduced negatives were made, and the etching of the copper-plated Duroid circuit boards has started. The art boards for the power splitter are about half completed.

The tentative decision has been made that the phased-array system be supported inside the spacecraft thrust tube by a pair of crossed structural members as shown in Figure 4-12. Since the major part of the phased-array



weight is in the phase shifter field coil and ferrite, the structure is designed to give primary support to them. The ground planes are then supported by the phase shifters and need not carry much weight.

The first set of 16 broad-band antenna elements have been returned to the shop to add provision of three tuning screws in the support tube as shown in Figure 4-13. Together with three quarter-wave matching sections inside the coaxial support tube, the input voltage standing wave ratio was better than 1.5 over the band from 3975 to 4200 mc. The screws were added to compensate for variations in length of the teflon spacers between the quarter-wave sections, and in the diameters of these sections. It is hoped that use of Rexolite spacers will eventually eliminate the need for these adjustments.

## PHASED-ARRAY CONTROL ELECTRONICS (PACE)

### Status

Preliminary circuit releases are now complete and final releases are 40 percent complete. Several logic changes have been made to incorporate the following features:

- 1) Operation during eclipse — provisions have been included so that execute pulses from the command system replace the solar sensor pulses during eclipse periods.
- 2) The  $\psi_2$  counter has been modified so the measurement error in counting the  $\psi - \psi_2$  angle is reduced from  $\pm 0.35$  to  $\pm 0.044$  degree.

Power requirements of the power amplifier unit have been calculated, both in normal and failure mode operation. Details of regulator operation in failure modes are being worked out with the regulator design personnel.

Figure 4-14 shows an oscilloscope trace of one of the special waveform generators.

### Jet Control Electronics (JCE)

A preliminary design review was presented during this period. The main purpose of the review was to discuss the various methods by which the control logic could be implemented. The recommendations indicated that the command decoder and jet control electronics should be intraconnected within a quadrant rather than interconnecting all command decoders to all jet control electronics. The preferred method results in seven less interconnected wires per quadrant and approximately 30 percent fewer parts per quadrant.

The final logic and circuits was released for product design during this period.

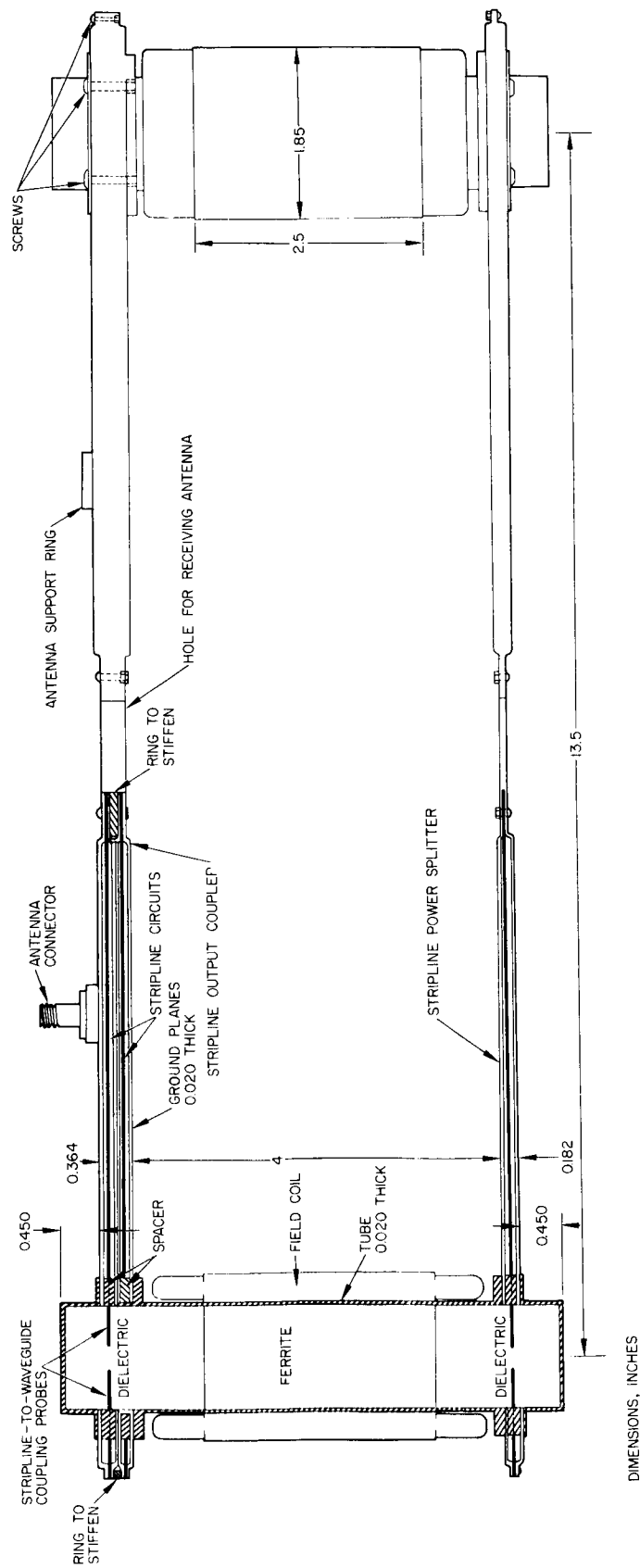


Figure 4-11. Phased Array Antenna, Cross Section of Phase Shifters

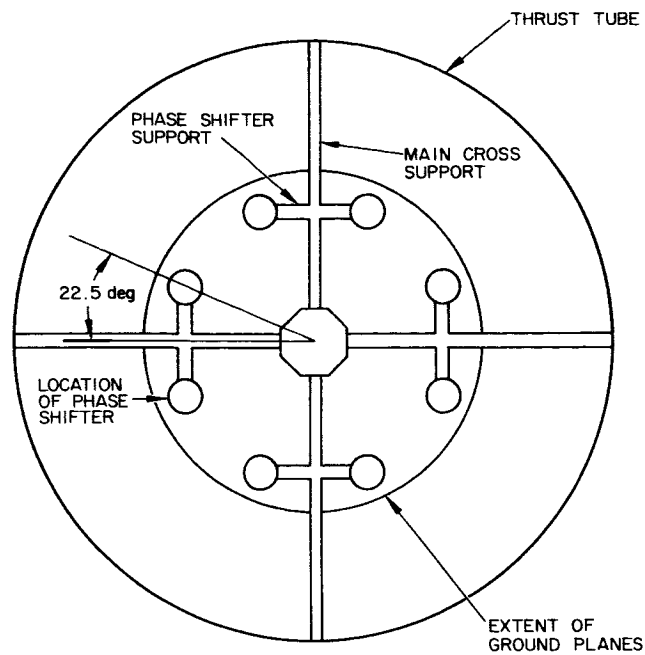


Figure 4-12. Support of Phased Array in Spacecraft

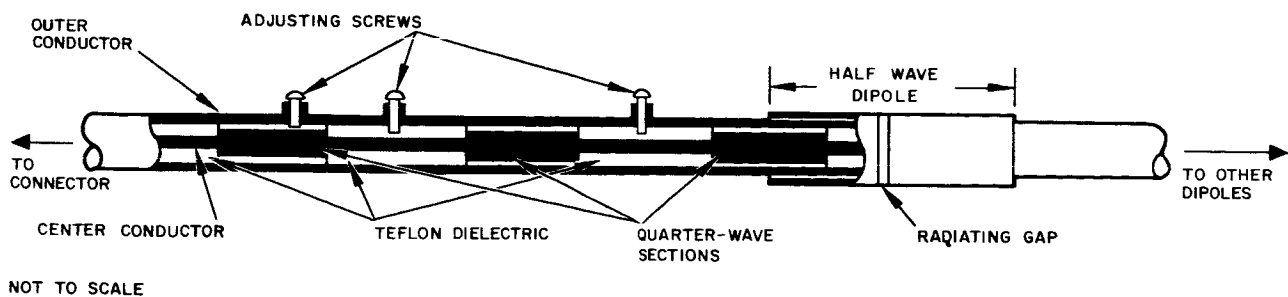


Figure 4-13. Phased Array Antenna Matching Arrangement (Cross section)

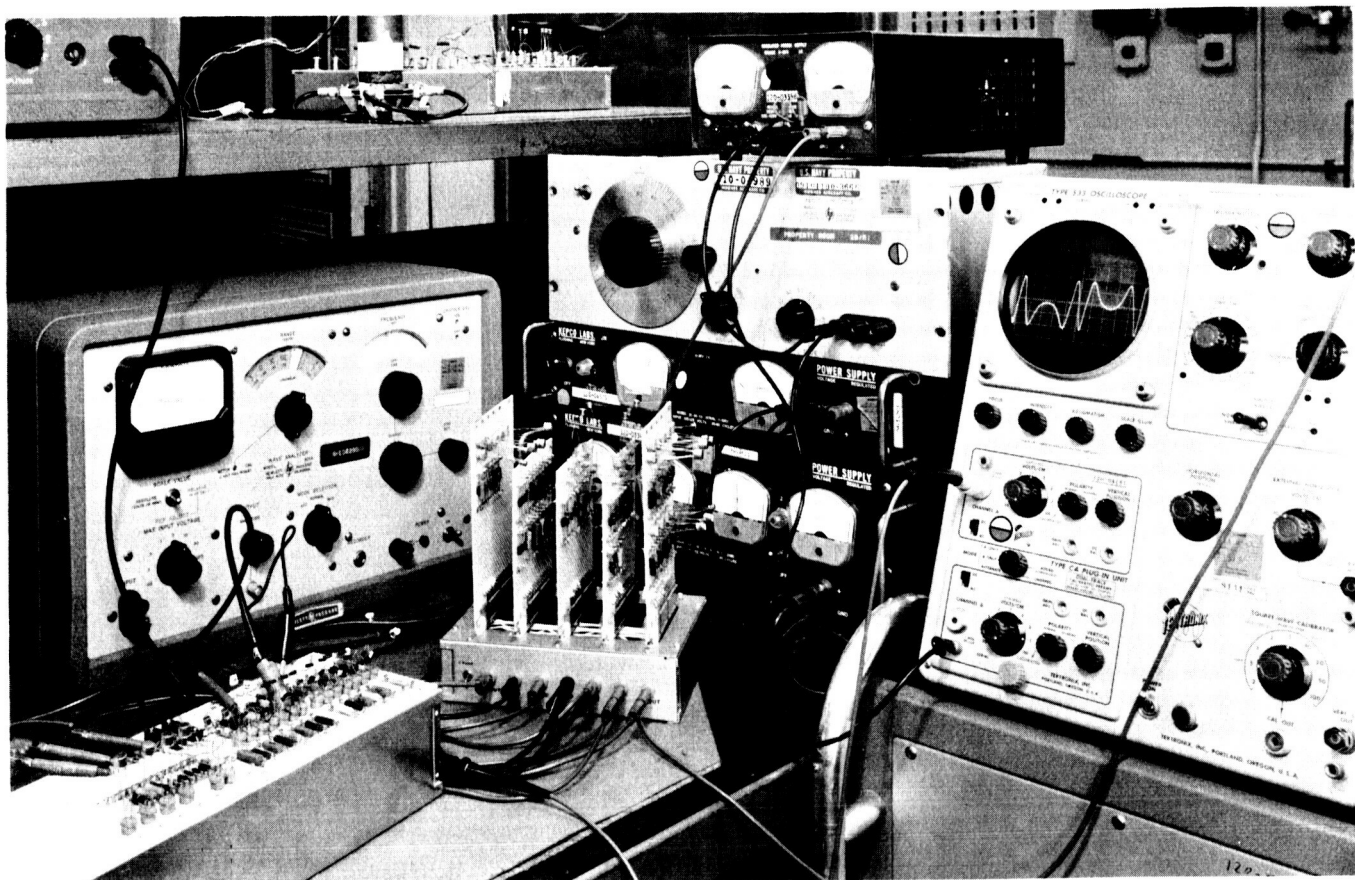


Figure 4-14. Test Setup and  
Waveform Trace One of Special  
Waveform Generator

A preliminary release of the solenoid amplifier is expected during the first portion of the next reporting period. This release will reflect final configuration. A coordination meeting was held with Marquardt Corporation to discuss the present electrical characteristics of the solenoid coil. Preliminary data was obtained and a data format for future tests was recommended.

Of the 37 card types required for the PACE/JCE advanced engineering model, 34 have been designed. The unit has been designed and drawings are being checked. The three other card types are being held for circuit releases. All semiconductors are on order. Fabrication will begin during the next report period.

Sixty-five percent of the spacecraft modules are in various stages of design and drafting. Product design of the boards and unit is awaiting definition of the space envelope for this subsystem. Weight and volume estimates have been made for the PACE/JCE.

#### PACE Regulator (160 Unit)

This regulator must supply 200 ma at +24 volts and 100 ma at -24 volts.

To achieve good  $\pm 24$ -volt tracking, a dc-dc converter will be used to supply  $\pm 24$  volts from the same secondary winding. This will be less efficient than supplying the -24-volt requirements from a -24-volt series regulator directly. To avoid poor load regulation for both polarity outputs, the series regulator -24-volt sensing point will be moved from the series regulator output to the dc-dc converter -24-volt output. It has been demonstrated that this remote sensing is feasible, but stability considerations associated with the high-gain feedback loop will require special attention. For instance, the series regulator ac sensing will have to be retained at the series regulator output to avoid instability associated with transportation lag through the dc-dc converter.

The 160-unit regulator must also contain circuitry to ensure that it may be turned off in the event of any internal failure. A transistor will be placed in series with the series regulator control transistor. The ON and OFF command pulses will be used to drive this transistor to saturation or cutoff respectively, in addition to energizing or de-energizing the series regulator. Thus, an independent regulator cutoff will be provided.

#### Power Amplifier Regulator (163 Unit)

This regulator must be capable of supplying 250 ma at  $\pm 24$  volts and 5 ma at  $\pm 35$  volts. As in the PACE regulator (160 unit), tracking of the plus and minus voltage magnitudes is desirable. Therefore the -24-volt output will be taken from the same dc-dc converter secondary as the +24-volt output. The same converter transformer will also have a secondary to supply  $\pm 35$  volts. There will be two power amplifier regulators per spacecraft,

for redundancy. However, the +24-volt output of each regulator will be joined together through an OR gate to form a +24-volt nonredundant bus. Also, a -24-volt nonredundant bus will be supplied from the two regulators through an OR gate. To provide good load regulation, the series regulator dc sensing will be taken from the -24-volt output of the converter. The stability considerations discussed relative to the PACE regulator above will be applicable to the power amplifier regulator also.

## CENTRAL TIMING ELECTRONICS

### Central Timer

Since the last report, four "count of 8" scaler stages and one shaper stage have been extensively tested in the laboratory and found to operate satisfactorily over the temperature range of -55 to +95° C. Loss of power has been shown not to affect the stored count. Variation in the -12-volt supply of  $\pm 5$  volts has not altered operation of the scalers.

Preliminary design of the apogee timer output latch is complete and is presently being tested in the laboratory.

Preliminary release of all timer circuits (minus the apogee motor driver) will be made by 8 June 1963. Preliminary design review of the central timer is scheduled for 24 June 1963.

All components for the two timers in the engineering model have been ordered. Fabrication of the timer is scheduled to begin on 15 June 1963.

### Discussion of Variable Apogee Ignition Considerations

A study has been made to determine the circuit problems involved in providing a variable apogee motor timer. Three configurations appear practical, but all have some shortcoming, especially in the area of reliability. NASA indicates the fixed timer configuration should be pursued.

For purposes of monitoring the timer operation on the gantry, it appears desirable to drive the timer reset windings from Agena power. Then, by commanding the Agena power off, the central timer could be enabled and its outputs monitored on the gantry. Also, the Syncom II - Agena interface connection would provide a positive open at separation to start the apogee timer.

## COLLINEAR ARRAY (CLOVERLEAF) RECEIVING ANTENNA

An additional unit of the cloverleaf array is being fabricated. This unit is designed to operate at the new center frequency (6165 mc). This array will be completed and ready for testing early in the next report period.

## VELOCITY AND ORIENTATION CONTROL

### Results of Computer Analysis of Spacecraft Orientation Maneuver

The first series of IBM 7090 computations of the spacecraft orientation maneuver has been completed. These are shown as trajectories of the spacecraft spin axis in the vicinity of the North Pole on a polar projection of the unit sphere, Figure 4-15.

The calculated results show North Pole acquisition for several jet firing angles  $\psi$  (i. e., relative to the sun line) when assuming constant spin rate (no operating spin rate control and no spin disturbance) and no nutation damping. A typical initial orientation based on orbital studies was used. The basic parameters assumed are given in Figure 4-15.

Along each  $\psi$  trajectory are plotted numbers of jet pulses to reach a given position. Since there are 100 pulses per minute, "pulses/100" is the time in minutes from the start of the acquisition maneuver.

From Figure 4-15:

- 1) The sensitivity of final-position to firing-angle variations is about 1 deg/deg. This is a significant parameter since several system errors lead to firing angle error. For example: jet timing, or spacecraft initial position error. (An error in spin-axis to sun line angle leads to the computation of an incorrect firing angle.)
- 2) Approximately 12.6 minutes and 1260 pulses will be required for the orientation maneuver.

Since these particular computer results embody the same basic assumptions made in preliminary calculations, the agreement between the two sets of results is significant. This agreement is a verification of the assumption made in preliminary calculations that the vehicle spin axis and angular momentum vector are in near coincidence during the orientation maneuver. The IBM 7090 program is thus checked out for a portion of its design capability. (Further program modifications now have a check base involving a major portion of program capability.)

Further IBM program development to include damper and jet dynamics is in progress and sensitivity studies of other parameters (such as jet thrust misalignment) will be undertaken.

### Spin Rate Mechanism

Testing of the Hughes model spin rate control during the report period continued in the following test areas:

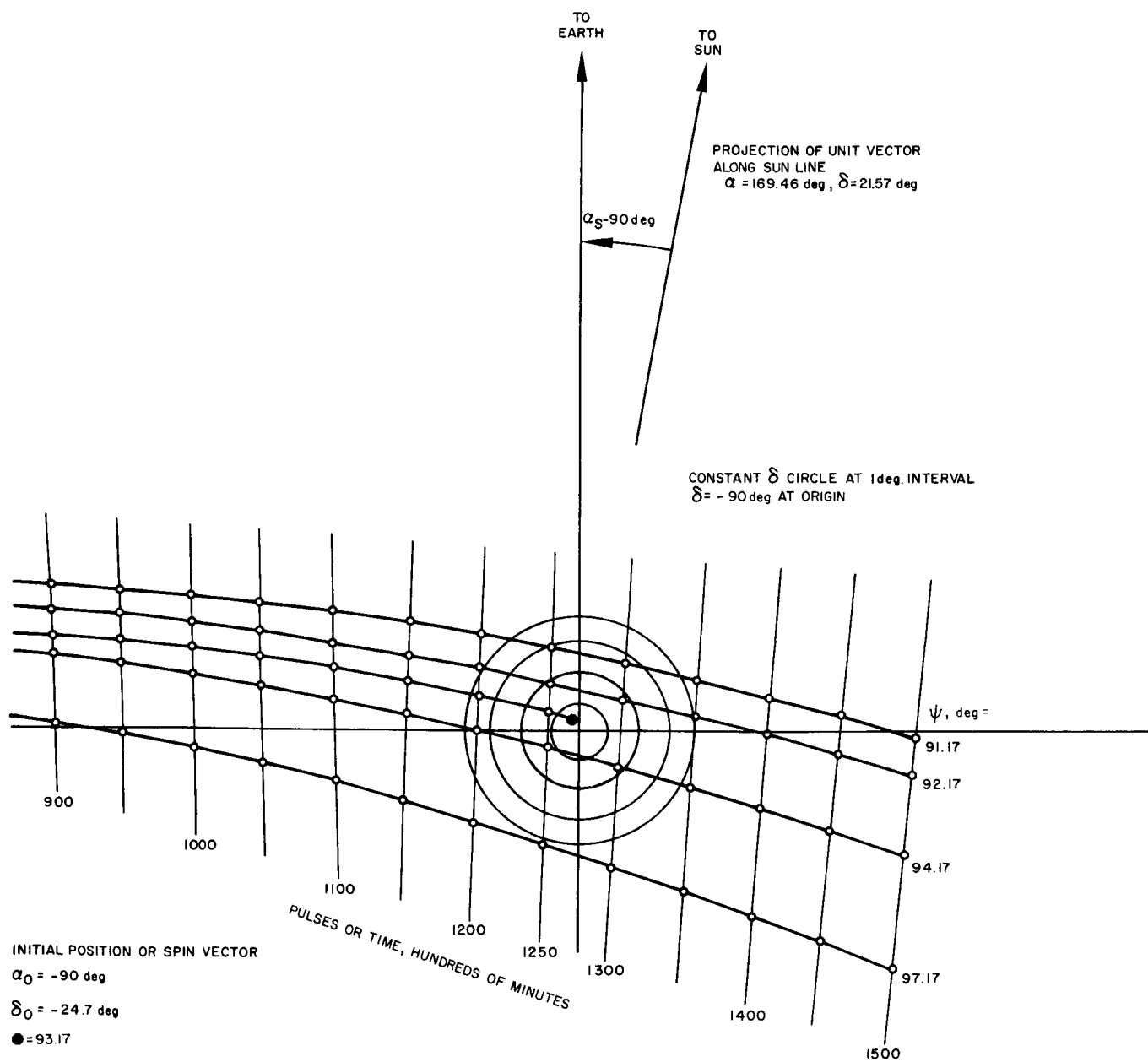


Figure 4-15. Mark II Spacecraft Orientation Maneuver



Transient decay

Vibration

Endurance

In addition, a refinement in damper-filling technique was investigated.

As a result of this month's effort and the present status of the Marquardt engineering model spin-rate control, further testing of the Hughes model is expected to be minimal, with emphasis largely on damper performance analysis rather than on both performance and durability.

#### Transient Decay Test

Accelerometer data on the transient decay performance of the Hughes model spin rate control, employing 2000 cs silicone fluid and an 0.040-inch-diameter orifice, is plotted in Figure 4-16. (Figures 4-17a and 4-17b are repeated from the April report for comparison.)

It is apparent that the decreased orifice size does not produce increased damping of the swung mass since the unit now requires more time to damp ( $\sim 1$  second) than before ( $< 0.5$  second). In addition, the nature of the response has been considerably changed.

However, using the results of this test in conjunction with those results from the 750 cs fluid and damper theory (refer to Advanced Syncom Summary Report, 31 March 1963), it is possible to estimate that the Hughes bellows damper design (Figure 4-17c) employing 750 cs fluid (and an 0.060-inch-diameter orifice) is capable of providing damping ratios near 0.50 at frequencies from 0 to 20 cps. This estimate is possible provided that an adequate filling of the unit with fluid is achieved, which is discussed below.

The transition of swiveled jet response from that of Figures 4-16 to 4-18 can be objectively discussed in terms of the following simple differential equation:

$$J s^2 \theta(s) + \frac{Ds\theta(s)}{\tau_D s + 1} + K\theta(s) = F(s)$$

where

$J$  = Moment of inertia of swung mass about pivot axis

$D$  = Damping coefficient =  $\rho A^2 R$

$\rho$  = Specific weight

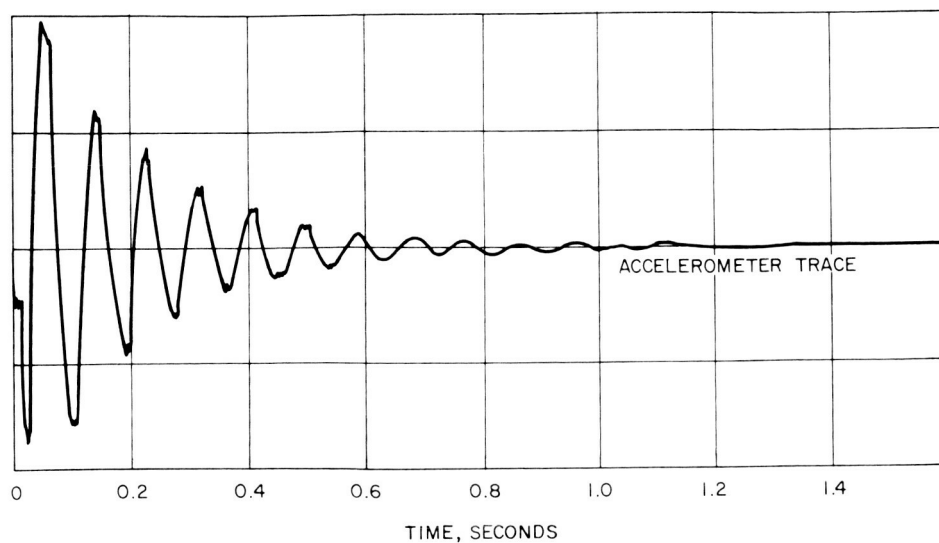
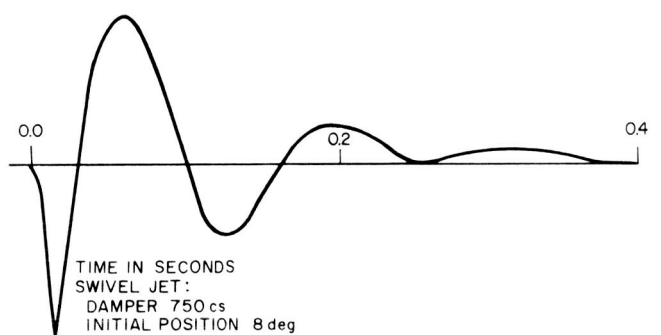
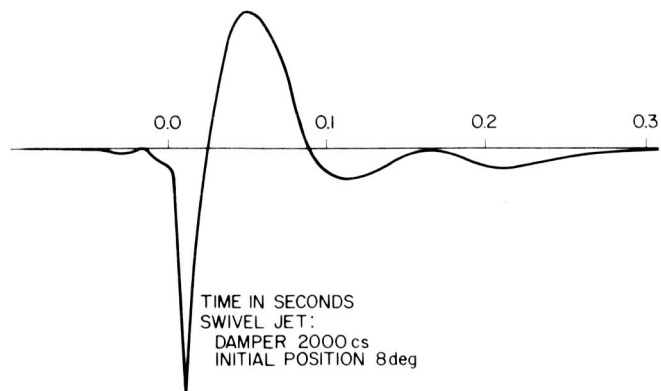


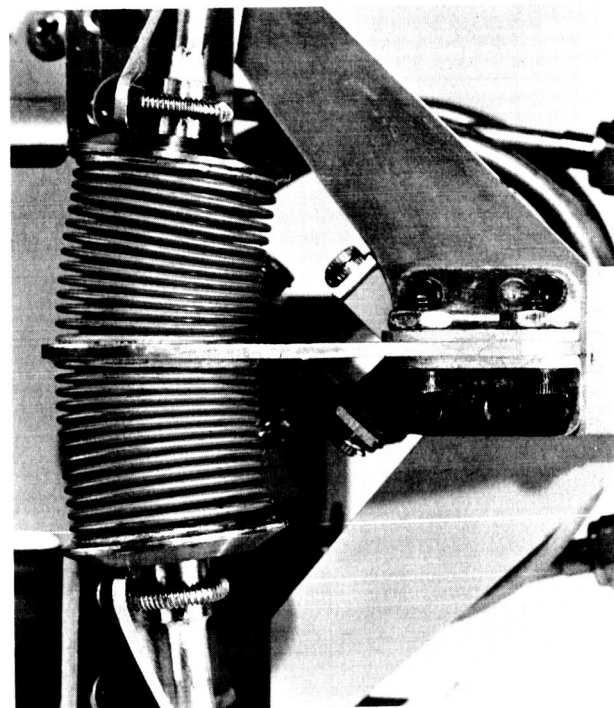
Figure 4-16. Mark II Hydraulic Damper



a) Transient decay of spin rate control for 750 centistoke silicon fluid



b) Transient decay of spin rate control for 2000 centistoke



c) Bellows damper

Figure 4-17. Syncom II Tests

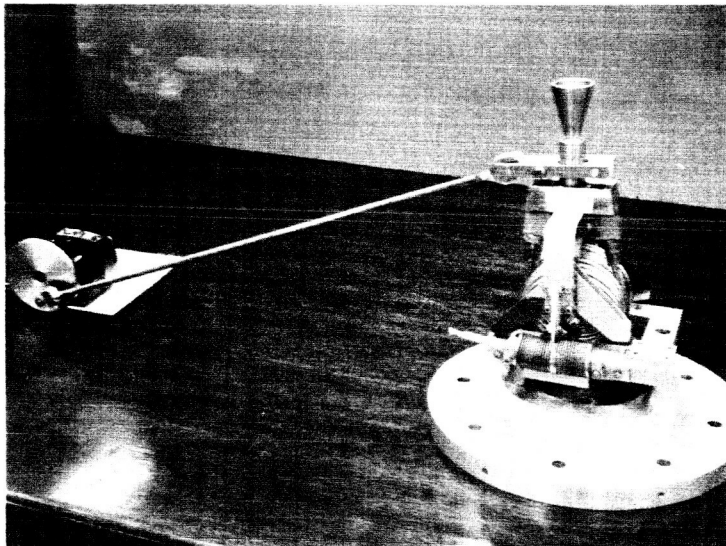


Figure 4-18. Spin Rate Control Endurance  
Test

A = Area of orifice plate subjected to fluid pressure

R = Hydraulic resistance of orifice

$\tau_D$  = Damper time constant =  $\frac{RC}{2}$

C = Hydraulic capacitance of one bellows chamber

K = Total static spring rate associated with pivoted mass

F = Forcing function

Considering the second term of the equation:

- 1) For low frequencies, damper action typical of viscous damping is expected.
- 2) For high frequencies, the damper should behave like an additional spring of spring constant  $2\rho A^2/C$ .
- 3) The dominant second-order frequency associated with the decay should increase with increased hydraulic resistance, the rest of the parameters remaining constant.
- 4) The damper time constant,  $\tau_D$ , determines which frequencies correspond to high and low.
- 5) The damper break frequency should be considerably greater than the undamped natural frequency. (In the Summary Report  $\tau_D \leq \frac{1}{3\omega_n}$  was estimated to give adequate damper performance.)

Table 4-1 compares some damper parameters.

Since hydraulic resistance (R) increases from top to bottom in the table, the observed frequency should, and does, increase also. Increasing R increases the damper time constant, shifting the undamped natural frequency toward the high-frequency range, thus increasing the effective system spring rate.

If the changed nature of the response as well as the shift in natural frequency is accounted for, it is reasonable to assume that test No. 4 is typical of high-frequency behavior. Assuming this and measuring the damping ratio for test No. 2 ( $\xi \sim 0.48$ ), it can be concluded that the response of the damper with 750 cs fluid is typical of the low-frequency response. More specifically, it can be shown, based on the damper equation, that

TABLE 4-1. DAMPER PARAMETERS

| Test Number | Orifice Size, inches | Fluid Viscosity, cs | Frequency, cps | Figure Number | Notes  |
|-------------|----------------------|---------------------|----------------|---------------|--|
| 1           | -                    | -                   | 7.0            | -             | Test No. 1 was performed with pinch tubes open; it corresponds to a no damping case. A check calculation of $\omega_n = \sqrt{K/J}$ gives $f_n = 7.1$ cps. |
| 2           | 0.060                | 750                 | 7.1            | 4-17b         |  |
| 3           | 0.060                | 2000                | 8.7            | 4-17c         |  |
| 4           | 0.040                | 2000                | 11.8           | 4-17a         |  |

$$\tau_D = \frac{1}{\omega_{no}} \left[ \frac{\xi \left( \frac{\omega_n}{\omega_{no}} \right)}{1 - \left( \frac{\omega_n}{\omega_{no}} \right)^2} \right]$$

if  $\omega_{no}$  is the measured frequency from the high-frequency case, and  $\xi$  and  $\omega_n$  are measured from the low-frequency case. Based on Table 4-1  $\tau_D = 6.15 \times 10^{-3}$  seconds. Now,  $1/3 \omega_n = 7.4 \times 10^{-3}$  seconds, so that typical low-frequency response from the damper of test No. 2 may be expected.

From the above analysis it may be concluded that a hydraulic bellows damper for the Syncom II application is feasible; the hydraulic resistance and capacitance needed can be made available in a unit of practical size and weight. Also, a basis for interpreting available spin rate performance data from the Marquardt unit has been established.

#### Vibration Test

Vibration testing of the Hughes spin rate control during May was carried on in two phases: Unit shake tests and T-1 shake tests.

The objectives of the unit shake tests were:

- 1) To check the feasibility of operating the spin rate control without pin pullers during booster rocket operation.
- 2) To check the durability of the bellows damper model.

These tests were performed (without pin puller) and recorded with a slip-sync camera. The procedure was:

- 1) Low-level (3-5 g) survey of three axes (parallel and perpendicular to the pivot axis) to determine the frequency bands of particular interest for filming at the higher vibration levels.
- 2) Double amplitude of 0.5 inch or 15 g sweeps in selected bands parallel to the pivot axis and the nominal position (zero degrees of swivel) of the thrust axis.
- 3) Double amplitude of 0.5 inch or 15 g sweep from 5 to 2000 cps in 4 minutes with forcing input perpendicular to both pivot and nominal thrust axis (worst direction from standpoint of operation without pin puller).

Observation of these tests and the motion pictures showed no outstanding oscillation of the damper unit, and no damage to the unit was found subsequent to the tests. However, it is concluded that because of the inability of the damper (2000 cs fluid and 0.060 orifice) to prevent violent oscillations of the pivoted mass, a pin puller will be required. (Note that the above tests were still not at the specification level of 50 g.)

The above conclusion may be modified if the amplification factors derived from the T-1 tests are much lower than expected. Reduction of the T-1 data will be completed during the next report period.

#### Endurance Test

The endurance test of the bellows damper was performed with the bellows damper in place on the Hughes spin rate control model. The damper was filled with 2000 cs fluid and employed an 0.040 orifice.

The model was cycled through its full travel (16 degrees, stop-to-stop) at approximately 1 cycle per minute by means of the apparatus in Figure 4-18. Examination of the damper post 5000 cycles showed no visible damage.

#### Damper Fill Technique

The technique outlined below has been evolved to permit a relatively high (a few microns) fill, while avoiding any possibility of gas entrapment by the entering fluid:

- 1) Seal one pinch tube.
- 2) Clamp damper in a vertical position as shown in Figure 4-19.
- 3) Connect damper to fill can with transparent tubing.
- 4) Set fill can to provide a few inches negative head on damper.
- 5) Pump apparatus down until fluid in fill can is quiescent at full vacuum.

- 6) Bleed in atmospheric pressure to force fluid into damper.
- 7) With fill can still attached, pinch aluminum tubing.
- 8) Removed fill can and seal pinched end.

#### Bipropellant Reaction Control System

Marquardt performed the first hot firing test using MON-15 and MMH on 3 May 1963. Tests were conducted in both a one-sixth duty cycle (1 second ON and 5 seconds OFF) and a one-third duty cycle (1 second ON and 3 seconds OFF). Test results indicate that nucleate boiling occurred in the oxidizer (MON-15) passage of the injector during the one-sixth duty cycle tests. Pulse mode tests were repeated with  $N_2O_4$  and MMH to determine whether the boiling phenomenon could be attributed to the higher vapor pressure of the MON-15. However, oxidizer boiling also occurred with  $N_2O_4$ . It was concluded that the more severe temperature soak-back conditions associated with a pulse mode operation resulted in oxidizer vapor pressures that exceeded injector pressure. Tests have been run on an injector design that displays an insulated insert to minimize heat input to the oxidizer. Preliminary test results are encouraging in that the oxidizer did not boil at the same injector head temperatures previously causing the problem. Additional tests of the insulated configuration are in process.

A flight-weight tank was fabricated and subjected to a hydro-burst test. The burst pressure of the tank was 1175 psi; design burst was 1180 psi. The tank exhibited excellent failure characteristics; the failure line crossed two welds without progressing along a weld line. Three other flight tanks have been fabricated, one of which was rejected because of inferior welding.

A major milestone was attained with the firing of the reaction control system in a feasibility demonstration on 24 May 1963. The unit is shown mounted on a spin table in the altitude chamber in Figure 4-20. The propulsion unit was rotated at a nominal 100 rpm, while the axial and radial engines were fired. The radial engine was fired for 15 minutes on a one-sixth duty cycle. The axial engine burned for 7-1/2 minutes on a one-third duty cycle. The position of the swivel-mounted, axial engine was monitored during the test. The mount reached the design positions at the required speed, indicating that spin control will be achieved. The Marquardt report for this system follows.

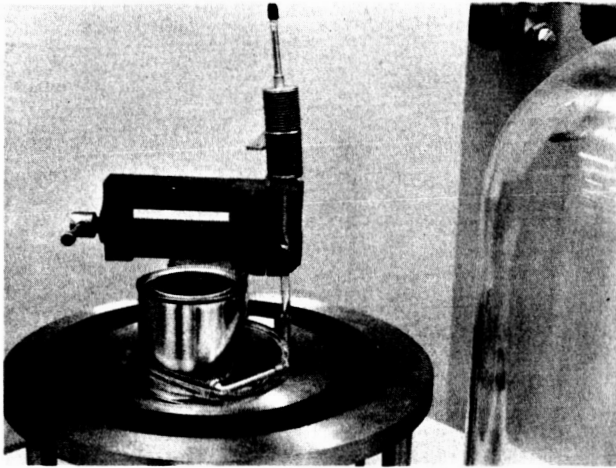
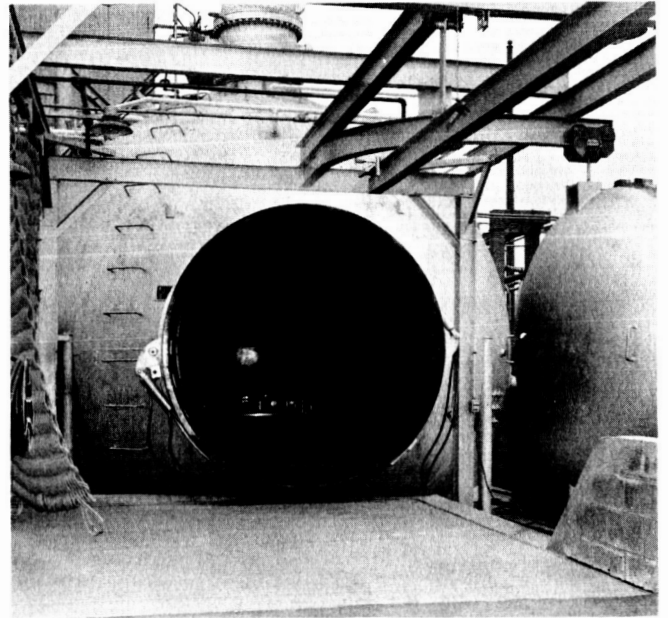
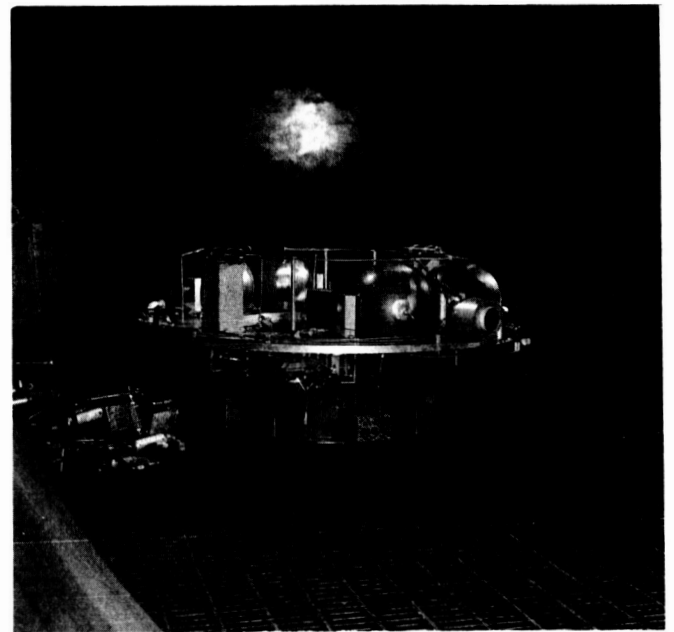


Figure 4-19. Damper Fill Apparatus



a) Exterior view



b) Interior view

Figure 4-20. Reaction Control System in Altitude Chamber



Marquardt Report MR-1-3 (Hughes - Edited)

SECTION I. ENGINEERING

DESIGN

SUMMARY

This report covers design support activity in the following areas:

1. ENGINEERING DRAWING RELEASE AND MAINTENANCE

Engineering Model assembly (upgrade)  
Engineering Model details (quick disconnect bracket assembly  
fixed restrictors - upgrade)  
System electrical schematic  
Rocket engine assembly  
Combustion Chamber (upgrade)  
Spin Control Mount  
Spin Control Mount and Engine Assembly

2. DEVELOPMENT TEST SUPPORT

Rocket engine tests (preparation of EEO's, TWEO's and Shop/Test  
Liaison).  
Flight tank (detailed development test plan).  
Long term storage (development test plan).

3. DESIGN REVIEWS

4. MANUFACTURING SUPPORT

DETAILED DISCUSSION

1. Drawing changes and maintenance were accomplished on the following:

Engineering Model Assembly - Drawing was changed to include various design details and notes regarding plumbing, fittings and

components. Final refinements are currently in work on this drawing. An electrical schematic of the engineering model was generated and will be released during the next reporting period.

Rocket Engine Assembly – The combustion chamber detail drawings (sea level and altitude) were changed to include design refinements improving the reliability of chamber raw material machining and coating.

Spin Control Mount – The spin control mount and engine assembly drawing was completed and is being checked currently. Release of this drawing will occur during the next reporting period. Minor changes were made to the spin control mount drawing to improve clarity and correct dimensional errors.

2. Development Test Support – Several experimental engineering orders and TWEO's were prepared and released in support of rocket engine tests. In addition, shop and test liaison support was expended as required to expedite delivery of test engines. A detailed development test plan is currently being generated for structural testing of flight type propellant tanks. A long term storage test plan is also in process.
3. Design Reviews – A design review on breadboard and engineering model components was initiated and will be completed during the next reporting period.
4. Manufacturing Support – Engineering effort in support of Manufacturing activities was expended on the following:

Tanks (Flight Type)  
Development Engines  
Components (Outside Manufacturer)  
Injector Valves (Flight Type)

## DEVELOPMENT

## SUMMARY

All engine tests conducted during the report period were made at altitude conditions. A combustion chamber burnout after a 24-second continuous run indicated that combustion chamber temperatures were higher than the temperatures experienced during sea level testing. Higher combustion chamber temperatures were verified during runs with thermocouples attached to the

combustion chamber wall. Tests made at a pulse duty cycle of 1 second on, 2 seconds off, showed that the engine could be run continuously at this duty cycle without the combustion chamber temperature exceeding 2800° F.

The oxidizer was changed from  $N_2O_4$  to MON-15 which required a change in the flow measurement techniques because the dark color of the MON-15 made it impossible to see the Rotameter float. A sight tube system was devised to obtain volumetric measurements of both the fuel and oxidizer flows and is satisfactory for measuring the flow for pulse testing and steady state testing up to 5 seconds. Continuous pulse duty cycle testing was conducted at duty cycles of 1 second on - 2 seconds off - and 0.1 second on - 0.5 second off at the maximum and minimum thrust levels with the MON-15-MMH propellant combination. At the low thrust condition a change in pulse characteristics (drop off in impulse) was observed as the engine temperatures increased. The change in performance is attributed to the combination of low oxidizer injection pressure at the low thrust condition and the increase in head temperature which approaches the vapor pressure condition for the MON-15.

An injector head modification to increase oxidizer and fuel injection pressures is presently being tested. Results of the testing has not yet been evaluated.

A summary of tests during the report period is shown in Table 4-2.

#### ENGINE TESTS 9a AND 9b

##### Test Objectives:

- 1) To determine engine performance at altitude conditions.
- 2) To demonstrate engine life capability.

##### Results:

Test 9a Engine runs of approximately fifteen seconds duration were made at altitude conditions to establish the desired fuel and oxidizer flow rates. Additional runs indicated that engine performance was not consistent for fixed propellant tank pressures and the test was terminated. A visual inspection of the engine showed that the oxidizer insert in the injector head had come out during the engine testing.

The engine was disassembled and rebuilt with X19151 S/N 003 injector head.

Test 9b. An initial fifteen second duration run at altitude conditions was made to verify the propellant flow rates. An endurance run was initiated, but terminated after twenty four seconds because of a drop off in engine thrust.

TABLE 4-2. SUMMARY OF TESTS

| Test Number | Test Date | Injector Head    | Combustion Chamber    | Test Conditions | Number of Runs | Total Burn Time | Maximum Single Burn Time | Remarks  |
|-------------|-----------|------------------|-----------------------|-----------------|----------------|-----------------|--------------------------|--|
| 9a          | 4/19/63   | X19151<br>S/N002 | X19158-501<br>S/N005  | Altitude        | 10             | 150 sec.        | 15.5 sec.                | Test terminated because of inconsistent engine performance |
| 9b          | 4/19/63   | X19151<br>S/N003 | X19158-501<br>S/N 005 | Altitude        | 2              | 39 sec.         | 24.0 sec.                | Chamber burn-out   |
| 10          | 4/25/63   | X19151<br>S/N003 | X19158-501<br>S/N 004 | Altitude        | 10             | 325 sec.        | 106.0 sec.               | Test objective completed                                   |
| 11          | 4/27/63   | X19151<br>S/N001 | X19158-501<br>S/N 004 | Altitude        | 8              | 105 sec.        | 9.0 sec.                 | Test terminated because of inconsistent engine performance |
| 12          | 4/28/63   | X19151<br>S/N002 | X19158-501<br>S/N 001 | Altitude        | 11             | 197 sec.        | 15.3 sec.                | Test objective completed                                   |
| 13          | 5/3/63    | X19151<br>S/N003 | X19158-501<br>S/N 004 | Altitude        | 11             | 1105 sec.       | 48.0 sec.                | Test objective completed                                   |
| 14          | 5/7/63    | X19151<br>S/N003 | X19158-501<br>S/N 004 | Altitude        | 75             | 325 sec.        | 3.0 sec.                 | Test objective completed                                   |

Inspection of the engine after the second run showed a burnout in the combustion chamber as shown in the photograph of Figure 4-21.

The appearance of the combustion chamber at the conclusion of Test 9b indicated high combustion chamber temperatures. Action was taken to install thermocouples on an altitude combustion chamber for subsequent engine testing.

## ENGINE TEST 10

### Test Objectives:

- 1) Obtain maximum combustion chamber temperatures at various combustion pressure levels.
- 2) Obtain combustion chamber temperature distribution.

### Results:

A total of ten runs were made to determine the nozzle temperatures as a function of O/F ratio and chamber pressure. A nozzle temperature of 2800° F, maximum, was used as a run shut down criteria to insure that the nozzle would not be damaged during the tests. The tests indicated that the combustion chamber temperatures were exceeding the 2800° F in less than ten seconds with the highest nozzle temperatures in the location of the previous combustion chamber burnout. At the low thrust condition ( $F_t = 3.2$  pounds) and  $O/F = 1.6$ , the combustion chamber temperature increased to 2800° F in approximately 17 seconds with the hottest area in the location of the previous combustion chamber burnout (half-way down the chamber on the fuel injector side of the engine). At high O/F ratios ( $O/F \cong 1.8$ ) at the 3.2 pound thrust, the combustion chamber temperatures were high with the highest temperatures on the oxidizer side of the engine. At the low O/F ratios ( $O/F \cong 1.27$ ) and 3.2 pound thrust, the combustion chamber temperatures were low permitting continuous engine operation without exceeding 2800° F.

A test at the high thrust condition ( $F_t = 4.8$  pounds) and low O/F ratio ( $O/F \cong 1.33$ ) showed the combustion chamber temperatures exceeded 2800° F in less than 10 seconds.

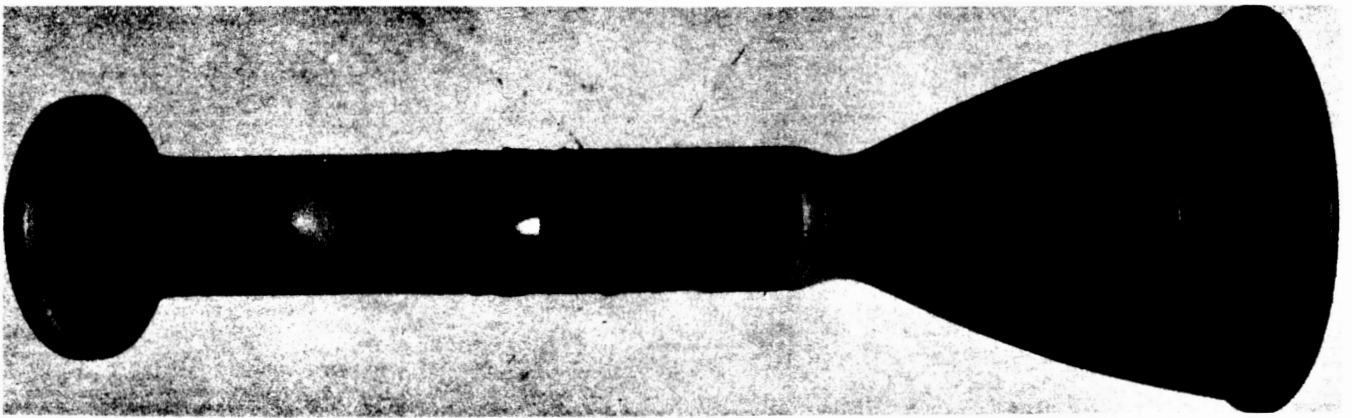


Figure 4-21. Syncom 5-1b Thrust Rocket Motor

## ENGINE TEST 11

### Test Objectives:

- 1) To determine engine performance and combustion chamber pressures with a modified injector head S/N 001.
- 2) To establish a pulse duty cycle to permit engine operation at a 5 pound thrust level at an O/F ratio of 1.62 such that combustion chamber pressures do not exceed 2800° F.

### Results:

The injector head used in this test differed from the injector heads used in previous tests in that it had a longer distance of the propellant impingement point from the injector face.

Short duration runs at the 3.3 and 5.0 pound thrust levels showed that engine performance was good. However, no decrease in combustion chamber temperatures in relation to the previous injector head configuration was observed.

Duty cycle operation (one second on, three seconds off) was initiated at the 5 pound thrust level. However, a decrease in performance and temperatures was observed shortly after starting the pulse duty cycle run. Subsequent investigation showed that the fuel insert head came out of the injector head.

Action was taken to provide positive retention of the injector inserts by staking. The configuration of S/N 001 injector head did not permit staking without damage to the injector inserts.

## ENGINE TEST 12

### Test Objectives:

- 1) To establish a pulse duty cycle to permit engine operation at a 5 pound thrust level and O/F ratio of 1.62 such that combustion chamber temperatures do not exceed 2800° F.

### Results:

The injector head used in this test (S/N 002) had a new oxidizer injector insert installed and both inserts were staked.

Short duration runs were made to establish the correct propellant flow conditions and runs were made at pulse duty cycles of one-third (1 second on, 2 seconds off), one-half (1 second on, 1 second off), one-tenth (1 second on, 9 seconds off) and one-sixth (1 second on, 5 seconds off).

The results indicated that nozzle temperatures would not exceed 2800° F for continuous operation at a one-third duty cycle of 1 second on and 2 seconds off.

### ENGINE TEST 13

#### Test Objectives:

- 1) Determine engine performance characteristics with MON-15 as the oxidizer.
- 2) Document engine performance at maximum and minimum thrust at the following duty cycles.
  - a) One-third duty cycle (1 second on, 2 seconds off)
  - b) One-sixth duty cycle (0.1 second on, 0.5 second off)

#### Results:

With MON-15 in the facility system, the Rotometers used to measure oxidizer flow were not usable due to the dark green color of the MON-15 which made it impossible to see the Rotometer float. The oxidizer flow measuring system was changed to the orifice type flow meter with a  $\Delta P$  gage to measure the orifice pressure differential. Short 5- to 10-second runs, time limited by a maximum combustion chamber temperature of 2800° F, were made to establish the propellant flow conditions. However, flow conditions were not steady state at the end of the run due to increased oxidizer line volume as a result of incorporating the orifice type flowmeter. Pulse tests at a one-third duty cycle (1 second on, 2 seconds off) and a one-sixth duty cycle (0.1 second on, 0.5 second off) were made at two thrust levels. At the high thrust level (~ 5 pound thrust) pulse repeatability was good as shown in Figure 4-22. At the lower thrust level (~ 3.7 pound thrust) Figure 4-23, a considerable change in the impulse per pulse was observed until the engine temperatures became steady state.

To obtain better resolution of the pulse flow, the flow meters in the fuel and oxidizer systems were replaced with sight tubes so that the quantity of flow per pulse (or a series of pulses) could be measured directly.



## ENGINE TEST 14

### Test Objectives:

- 1) To determine pulse flow and engine performance with sight tube type flow meters with MON-15 oxidizer.
- 2) To determine pulse performance at low thrust levels (i. e. , low injector pressure levels).

### Results:

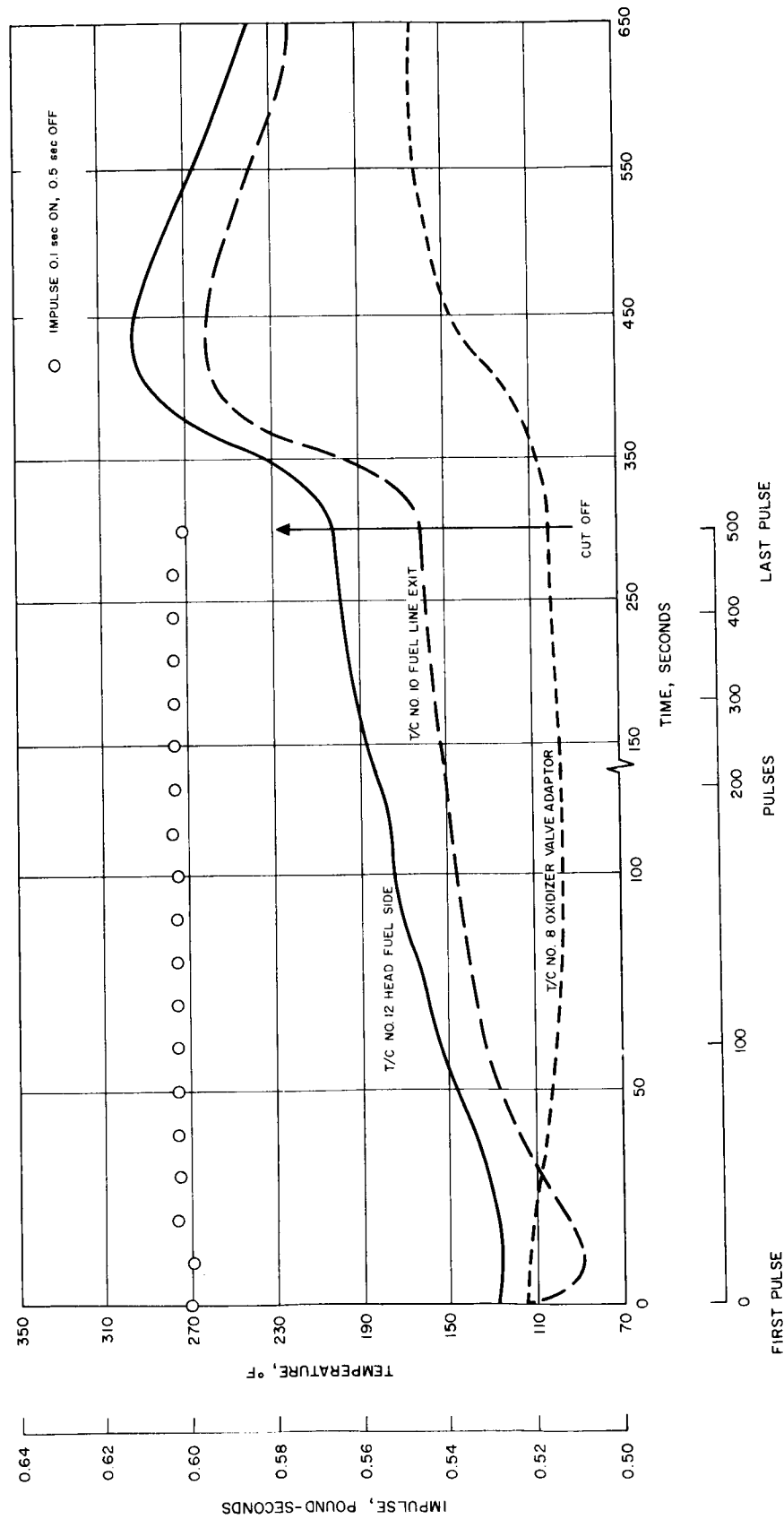
A series of pulse tests were conducted to determine the engine performance during pulse runs of 1-second duration by observing the quantity of propellant consumed on the sight tubes. The specific impulse for 1-second runs at the 5-pound thrust level,  $O/F = 1.67$ , was 266 seconds. A series of 1-second runs made at low injector pressure levels showed a decrease in oxidizer flow as the engine temperatures increased.

An injector head modification was instigated to increase the oxidizer injection pressures in order to prevent the oxidizer from reaching the vapor pressure condition at the low thrust levels as the head temperature increases.

Action was also taken to revise the facility plumbing to make it as close as possible to the vehicle system in order to obtain pulse dynamic conditions similar to vehicle operation.

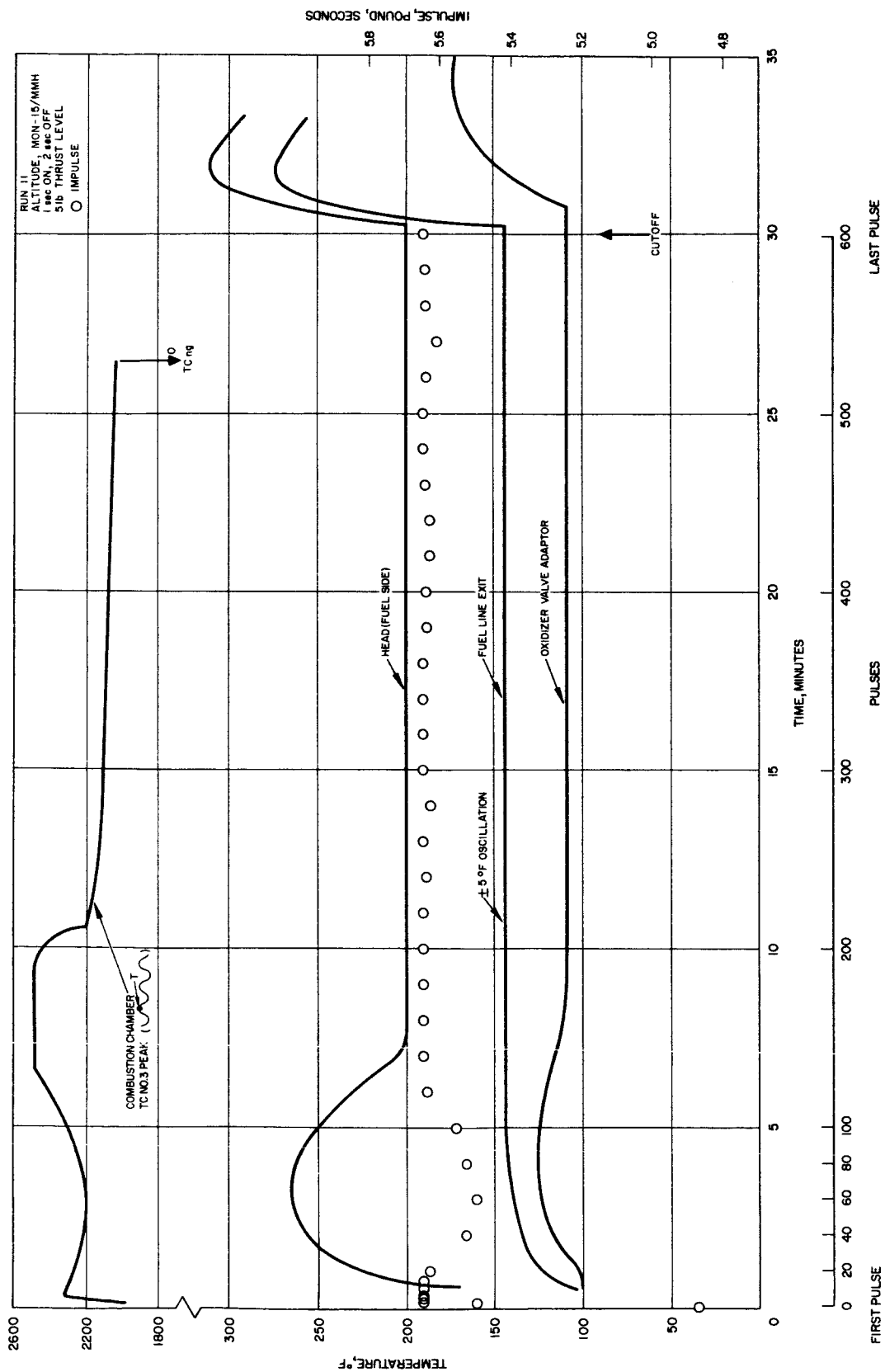
### ANALYSIS

A one-third duty cycle for the axial engines consisting of 1 second of operation followed by 2 seconds of cooling is currently being proposed by TMC to reduce the thrust chamber temperatures. For 1 second of operation the specific impulse is predicted to decrease by less than 1 percent compared to continuous operation. (See Section IV.) The prediction is based on extrapolation of pulse data for a 100-pound thrust engine using MMH and  $N_2O_4$ . Based on measured continuous performance at the 5-pound thrust and 3-pound thrust levels in altitude tests with MMH and  $N_2O_4$  the estimated overall mission specific impulse is 292 seconds. The overall mission is assumed to consist of 20 percent of the impulse obtained by 0.1-second pulses and 80 percent of the impulse obtained by 1-second pulses (see Section IV). The estimates will be made for MMH/MON-15 when performance is obtained in test. The effect of instrumentation accuracy on the estimated delivered mission specific impulse will be included when sufficient data is accumulated to allow estimation of thrust flow and impulse instrumentation accuracy.



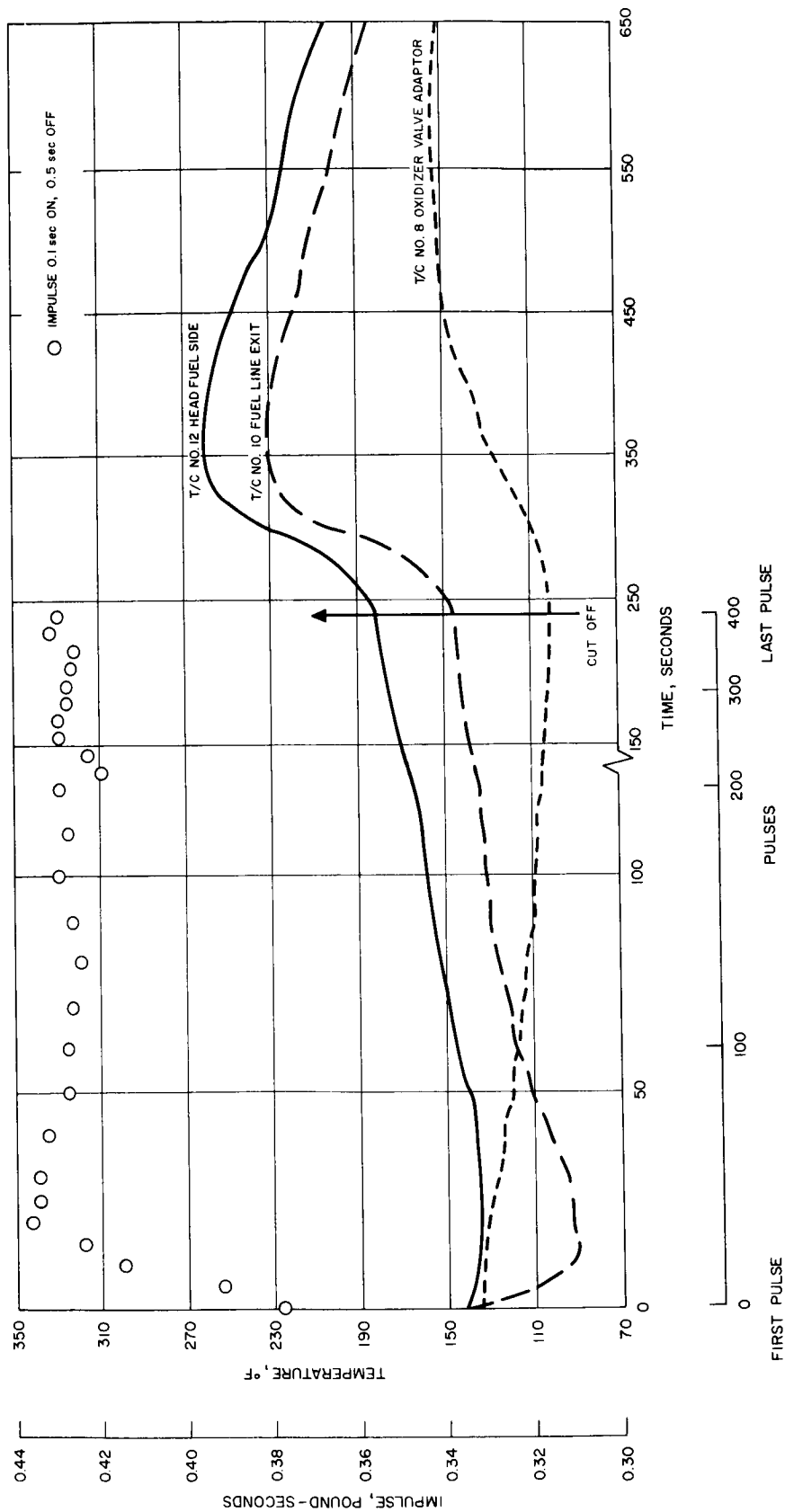
a) Run 10

Figure 4-22. Syncom Engine Temperatures and Impulse, 5-pound



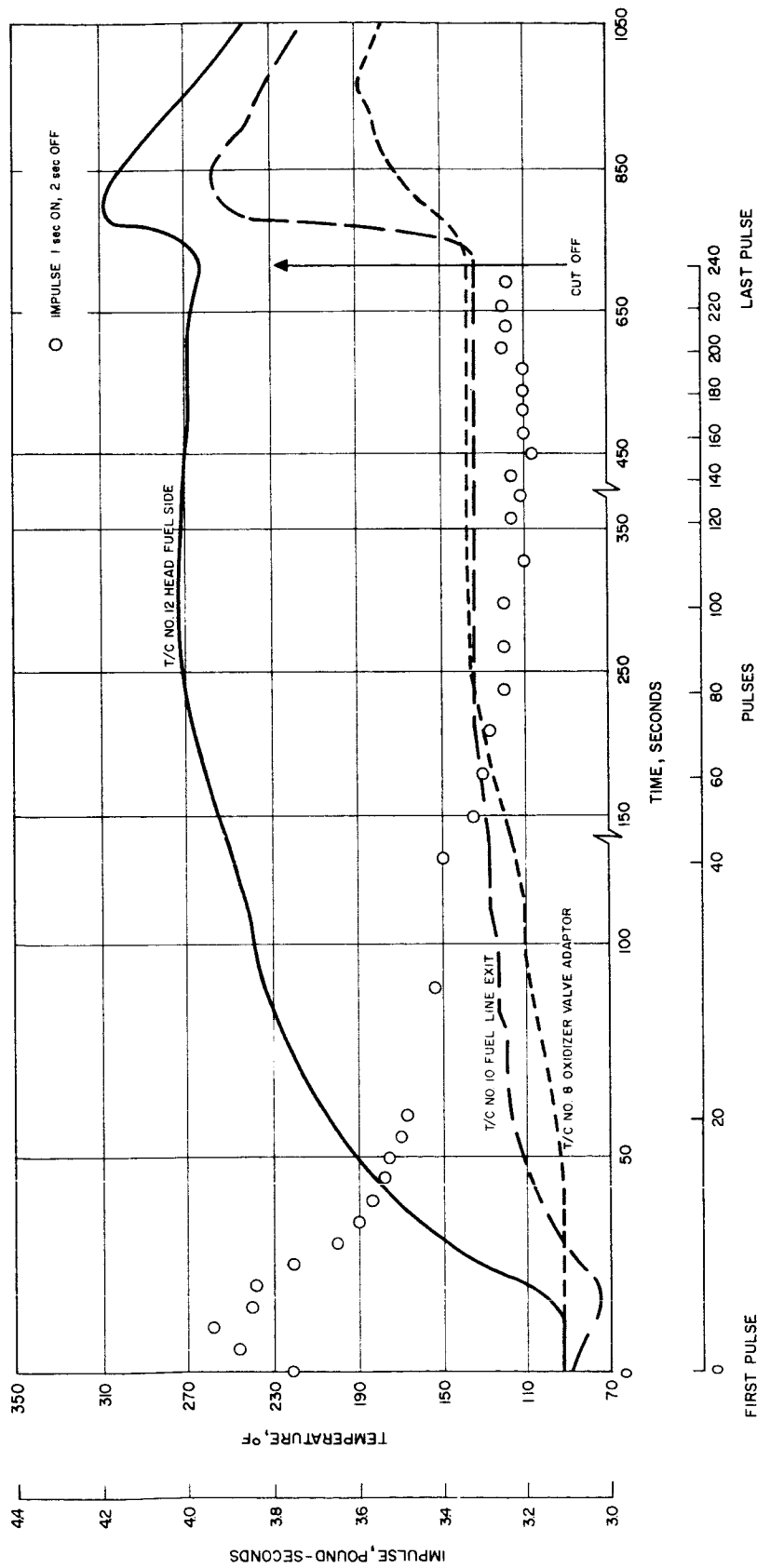
b) One-third duty cycle

Figure 4-22. Syncom Engine Temperatures and Impulse, 5-pound



a) Run 9

Figure 4-23. Syncom Engine Temperatures and Impulse, 3.7-pound



b) Run 8

Figure 4-23. Syncom Engine Temperatures and Impulse, 3.7-pound

Thermocouple measurements obtained in altitude tests indicate excessively high thrust chamber temperatures for continuous operation at the 5-pound thrust level. The temperature measurements indicate that the steady-state temperature at the 5-pound thrust level would be approximately 3300°F, which is approximately 550°F higher than indicated by optical pyrometer measurements obtained during long continuous runs at sea level at the same combustion chamber pressure. Thrust chamber life for a coated moly material system is inadequate at temperatures exceeding about 3000°F. A summary of the thermocouple measurements obtained in altitude tests of the 5-, 25- (Advent) and 100-pound (Apollo) engines is presented in Figure 4-24. The measured combustion temperature transients for the 5-pound engine are much steeper than for the 25- and 100-pound thrust engines. All three combustion chambers have the same contraction ratio and are approximately equal in length and wall thickness. The altitude thermocouple measurements for the 5-pound thrust engine are much higher than indicated by the optical pyrometer measurements obtained in sea level tests and also higher than indicated from calculated heat transfer coefficients obtained in transient temperature measurements on steel thrust chambers. Analysis of the results presents two major possibilities for the discrepancy between the sea level and altitude temperature measurements.

1. Forced convection in sea level tests around the outside of the thrust chamber due to jet pumping action of the exhaust greatly reduces the combustion chamber wall temperature. If so, the actual combustion gas to wall heat transfer coefficient is over two times as high as for the 25- and 100-pound thrust engines. If this is true, engine scaling from the 25- and 100-pound thrust engine size to the 5-pound thrust size is not valid. However, for both the 25- and 100-pound thrust engine, the temperatures measured in sea level tests essentially agree with the measured temperatures in altitude tests, although convection has a significant effect on the 100-pound engine moly flange and injector head temperatures.

2. The altitude facility and instrumentation used may not provide correct temperature measurements. For the 25- and 100-pound thrust engines, the engine exhaust acts as a no-flow ejector to aid in evacuating the cell and lowering cell pressure during operation. The small 5-pound engine does not provide no-flow ejector action. It is possible that exhaust gasses recirculate into the cell and convect and radiate to the thrust chamber. Theoretically, this affect on thrust chamber temperatures is second order compared to the input from the combustion gas to the wall. The 100-pound thrust engine cell contains a viewing port. When the no-flow ejector affect does not occur the 100-pound engine exhaust gas can be seen to recirculate and cause the platinum thermocouple wire to become incandescent, even though the combustion chamber temperature is known to be low. There is therefore the possibility that the high heat transfer coefficient to the small wire even at cell pressures of less than 0.1 psia result in conduction to the thermocouple junction resulting in an erroneous junction temperature and indicated combustion chamber temperature.

The cause of the high temperature indication during altitude tests has not been pin-pointed at this time. The engine test program is currently based on assuming the thermocouple measurements and test simulation to be a valid indication of actual combustion chamber temperatures.

Figure 4-25 presents test data of thrust chamber temperature versus duty cycle for altitude tests with MMH/N<sub>2</sub>O<sub>4</sub> for pulse widths of 1 second. The temperatures for the one-half duty cycle had not reached steady state after 19 pulses. The predicted peak temperature for the higher duty cycles shown is based on calculations using the available altitude temperature test data. The test data indicates that the maximum chamber temperatures for a one-third duty cycle are approximately 2500° F, which should allow for many hours of operational life. The maximum allowable pulse width at a one-third duty cycle is 3 seconds, based on the existing altitude temperature test data. For a duty cycle of 3 seconds "on" and 6 seconds "off", the peak chamber temperatures should reach 2860° F at the end of the pulse and cool to 1860° F at the end of 6 seconds (see Figure 4-26). The restriction on minimum pulse width for the axial engine is in maintaining a pulse efficiency as close to that for steady state operation as possible. If the pulse width is 0.5 second the loss in efficiency should be two times as great as for a 1-second pulse, or 1.5 to 2.0 percent less than for steady state.

## RELIABILITY

- 1.0 A revised Reliability Program Plan, modifying, and/or clarifying areas of activity has been completed.
- 2.0 A reliability block diagram has been completed.
- 3.0 A reliability apportionment study of components, subsystems, and system is in process.
- 4.0 A review of present Syncom polishing techniques on chambers is in process.
- 5.0 Releases processed through the group:
 

|                         |      |
|-------------------------|------|
| New drawings            | - 17 |
| Changed drawings        | - 6  |
| Reserve change EO's     | - 15 |
| Info and Instr EO's     | - 4  |
| Memo EO's               | - 7  |
| Supplemental EO's       | - 3  |
| New MTS                 | - 5  |
| MTS Reserve Change EO's | - 3  |
| New MTP                 | - 1  |
| New EPS                 | - 1  |

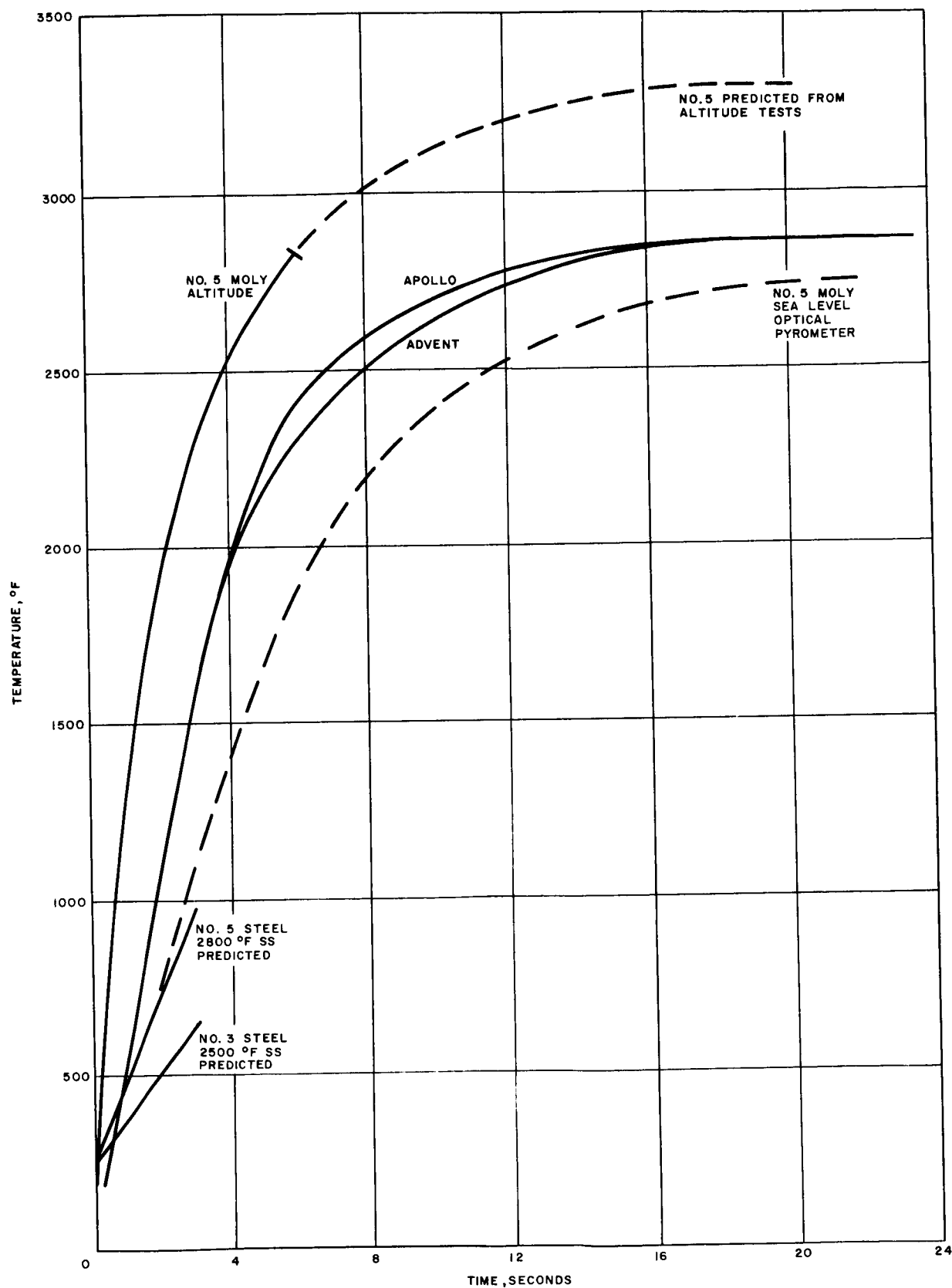


Figure 4-24. Thrust Chamber Wall Temperature



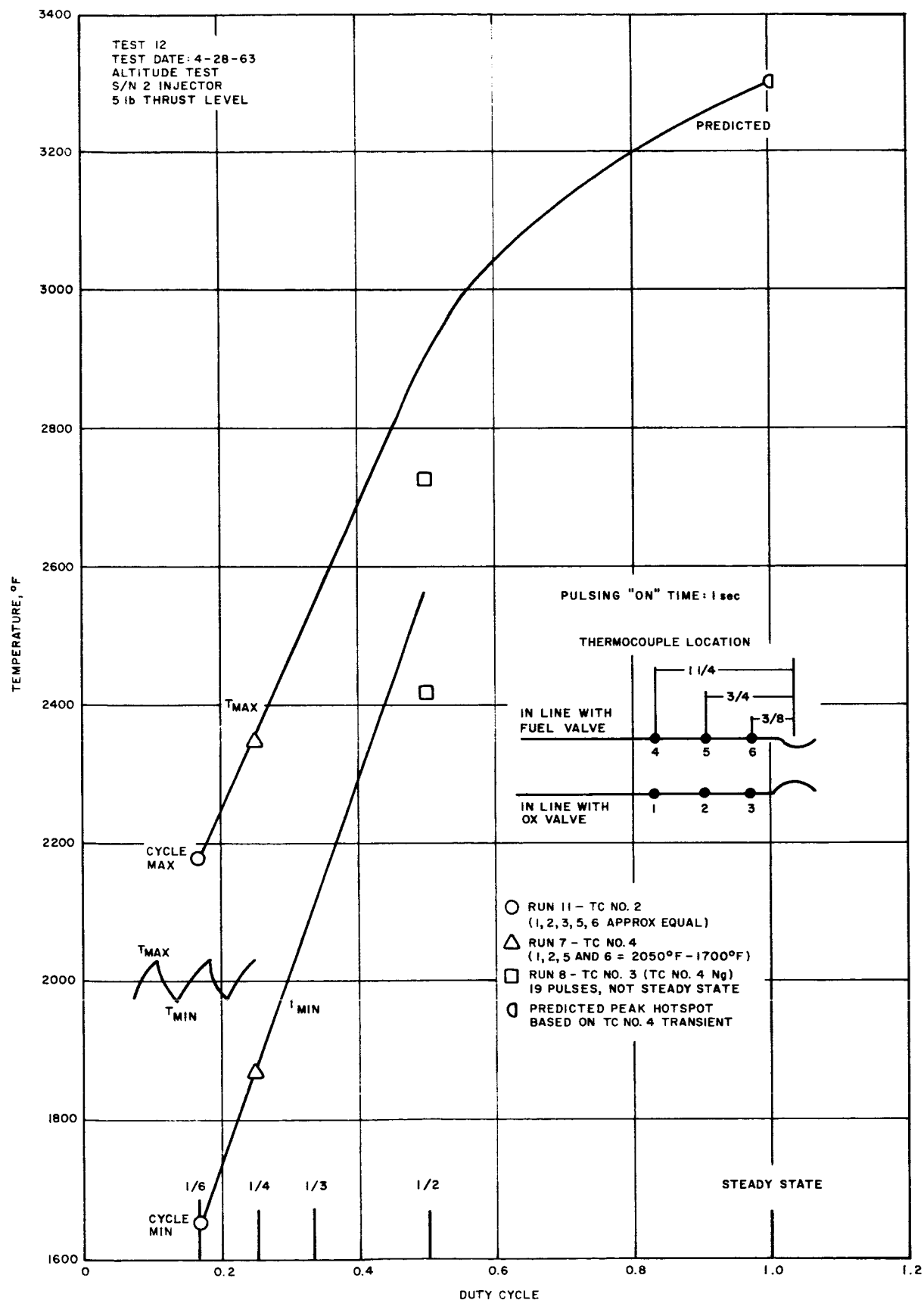
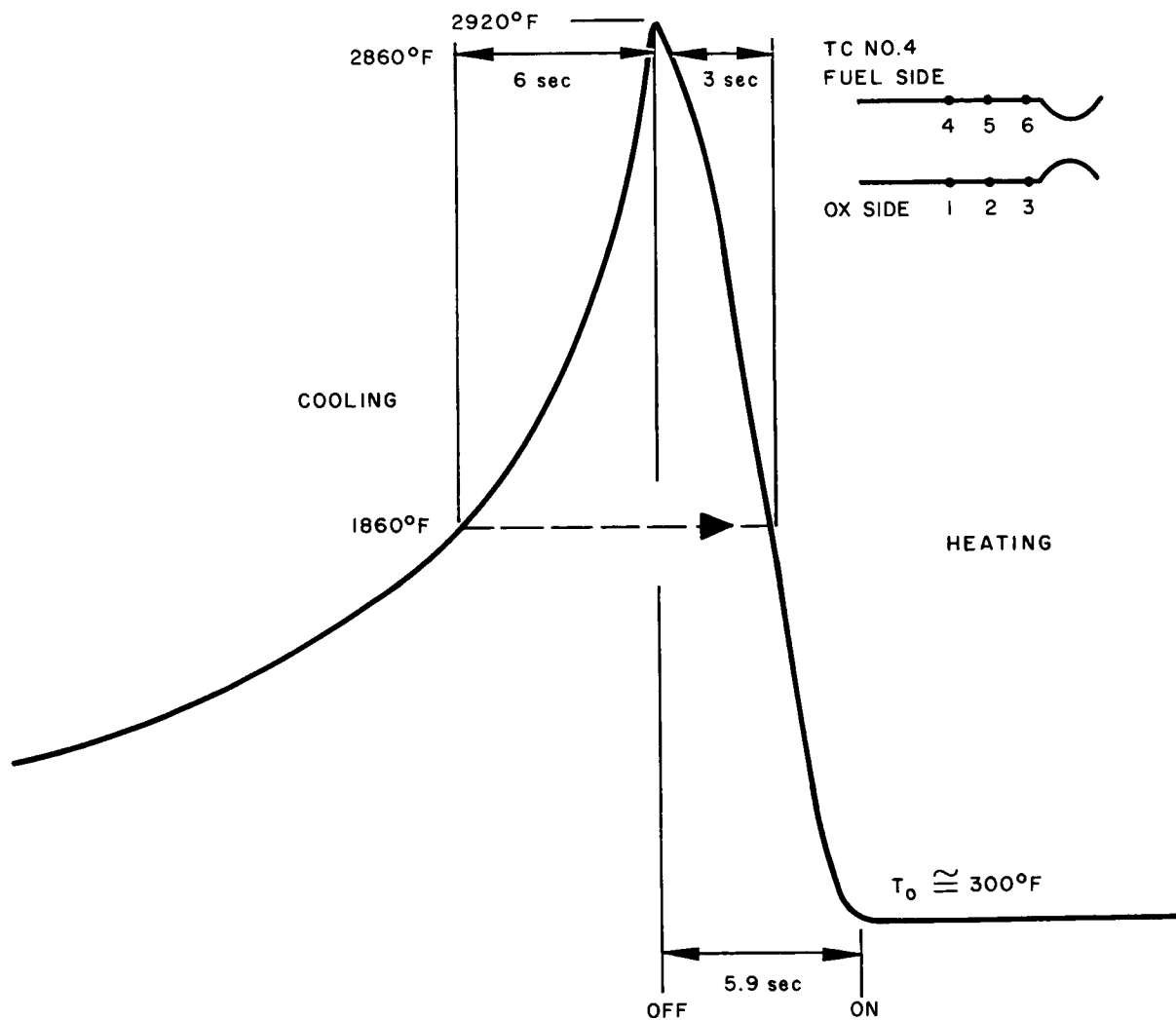


Figure 4-25. Thrust Chamber Temperature versus Duty Cycle  
4-55

FROM TEST 13, RUN 2  
ALTITUDE, MON-15/MMM



IF  $T = 2860^\circ\text{F}$  AT TIME OF SHUT DOWN, THE CHAMBER COOLS TO  $1860^\circ\text{F}$  AFTER 6 SECONDS. IF OPERATION IS INITIATED AT TC NO. 4 =  $1860^\circ\text{F}$ , IN 3 SECONDS THE PEAK TEMPERATURE WILL REACH  $2860^\circ\text{F}$ .

MAX ALLOWABLE ON TIME : 3.0 sec FOR 1/3 DUTY CYCLE

CHAMBER TEMPERATURE CYCLE AT TC NO. 4 LOCATION:  
 $2860^\circ\text{F}$  TO  $1860^\circ\text{F}$

Figure 4-26. Syncom Maximum Allowable "ON" Time for 1/3 Duty Cycle

## MATERIALS AND PROCESS

### Design Support

- 1) The TMC IR&D sponsored altitude chamber of 90Ta-10W has been manufactured and is scheduled to be coated with tin-aluminide by General Telephone and Electronics Research Laboratories at Bayside, Long Island. M&P will provide technical monitoring at GT&E.
- 2) Examination was made of altitude chamber X19158-501, S/N 005 after firing malfunction (19 April). The disilicide coating on chamber OD upstream of throat showed evidence of melting. Estimated temperature was approximately 3500° F.
- 3) A study was made for recommended system cleaning, decontamination and drying procedures. A procedure was recommended by IOM Ref: 154/58 dated 10 May 1963.

### Manufacturing Support

- 1) Technical monitoring of vendor disilicide coating process was conducted on altitude combustion chambers S/N 006 and S/N 007 P/N X19158-501. After the first of the two cycle coating operation, no crazing was observed. Slight crazing was seen after the second cycle before smoke testing. The smoke tests for coating continuity were satisfactory.
- 2) Conducted technical monitoring for preparation and heat treatment of propellant tank X19166 (T-tank). The subsequent burst pressure test required 1150 psi.
- 3) Conducted technical monitoring for preparation and heat treatment of X19181 bellows assembly (three units). The subsequent leak tests showed zero helium leakage on two units and a slight ( $1 \times 10^{-5}$  cc/sec helium) leakage on the third unit.
- 4) Conducted technical monitoring for preparation and brazing of injector head assembly X19151-501 (one unit).
- 5) Conducted technical monitoring for preparation and brazing of valve body X19176 (two units).

- 6) Conducted technical monitoring for heat treatment of mechanical test specimens X12821-501. These specimens were from sheet used to fabricate X19166-450 shell.

Quality Control Support

- 1) Conducted tensile tests on heat treated 17-7PH specimens for evaluation of raw material suitability for tank assembly X19166. Mechanical property test results showed all material to be acceptable. Specimens were heat treated by each TMC and Vac-Hyd Processing Corporation.
- 2) Provided technical evaluation for vendor survey on heat treatment of X19181 bellows assembly. Facilities surveyed included Vac-Hyd and Metallurgical Consultants Incorporated. MCI was eliminated due to faulty vacuum equipment.
- 3) Conducted coating thickness tests on control specimens representative of vendor coating of altitude chambers S/N 006 and S/N 007 (X19158-501). Results conformed to requirements:  $0.0020 \pm 0.0003$  inch.

## SECTION II. MANUFACTURING

### I. BREADBOARD ASSEMBLY

All motor components are complete and the balance of this assembly is complete.

### II. ENGINEERING MODEL

Approximately 95 percent of the bracketry and supports for the Hughes Aircraft Company structure is complete. The attach plate is reworked and ready for installation.

The X19166 Tanks are approximately 60 percent complete. Some porosity exceeding tolerance limits has been observed, but subsequent rework has eliminated the problems.

The motor components for the engineering model are approximately 80 percent complete, and the X19221 Heat Shield is approximately 60 percent complete.

### III. MANUFACTURING - GENERAL

All problem areas are well defined now. Using the same manufacturing process, some tearing of the bore on the X19158-501 Nozzle occurs sporadically, but this is to be expected and subsequent hand polishing removes the tear marks.

The first test tank (17-7 PH) was heat treated successfully.

### IV. O. P. COMPONENT STATUS

The Mount Assembly - Spin Control (X19194) and the Variable Restrictor (X19181) are the only undelivered O/P components at this date.

One unit of X19194 is scheduled for 15 May 1963 and the remaining two units are scheduled for 17 May 1963. All three X19181 units are scheduled 15 May 1963.

### SECTION III. TEST

#### FACILITIES AND STE

During the past period all remaining major facility and STE items were completed.

#### OPERATIONS

Testing in the ATL area continued. Difficulty has been encountered using rotometers due to the color (green-black) of the MON-15. Manometer type pulse flowmeters are being used to monitor pulse operation of the engine.

Cell 9 is complete and spin test of the breadboard will be started upon completion of feasibility testing and installation of the engines.

# SECTION IV. PREDICTED SPECIFIC IMPULSE FOR ONE SECOND PULSES AND FOR OVERALL MISSION

## Syncom Predicted $I_{SP}$ for 1 sec Pulse (Figure 4-27)

### 1. Dribble Volume (DV)

$$DV \equiv \frac{\text{Volume downstream of valve seat}}{\text{Volume of propellant for 10 ms steady flow}}$$

at O/F = 1.62

5-pound thrust:  $(DV)_{\text{fuel}} = (DV)_{\text{ox}} \simeq 50$  percent

Minimum thrust:  $(DV)_{\text{fuel}} = (DV)_{\text{ox}} \simeq 75$  percent

(100-pound thrust on Apollo  $(DV)_{\text{fuel}} = 58$  percent)  
 $(DV)_{\text{ox}} = 100$  percent)

### 2. Apollo Test Data

The following is test data for Apollo 100-pound engine

| Electrical<br>"on" Time,<br>milliseconds | $I_{SP}^*$<br>seconds |
|--|-----------------------|
| 30                                       | 214                   |
| 50                                       | 250                   |
| 100                                      | 275                   |

\*Corrected sea level data to altitude.

$I_{SP}$ , steady state (SS) = 300 seconds

$$I_{SP_{SS}} - I_{SP_{\text{pulse}_t}} = (I_{SP} - I_{SP_{C50_{ms}}}) \times \frac{50 \text{ (ms)}}{t \text{ (ms)}}$$

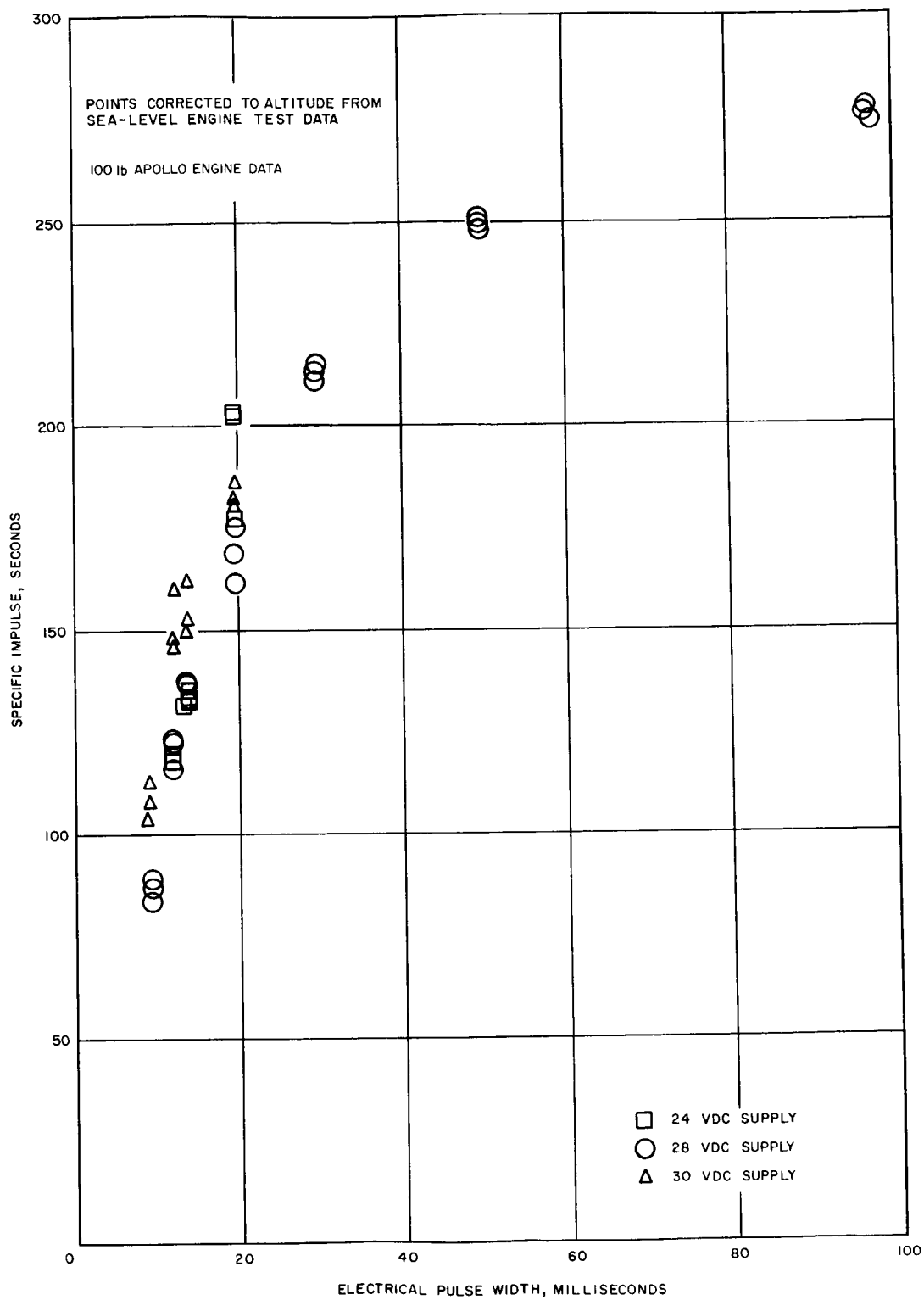


Figure 4-27. Specific Impulse versus Electrical Pulse Width



$$I_{SP_{pulse, t}} = I_{SP_{SS}} - (300-250) \frac{50}{t}$$

$$I_{SP_{pulse, t}} = I_{SP_{SS}} - \frac{2500}{t}$$

(4-1)

### 3. Estimated Performance of Syncom:

Equation 4-1 assumes that the startup and tail off conditions are independent of pulse width for a given engine.

Since the dribble volume of the Syncom engine is less than for the Apollo engine, the pulse performance of the Syncom engine compared to steady state  $I_{SP}$  should be higher than for the Apollo engine.\*

Extrapolating the Apollo data:

At 1 second (1000 ms), from Equation 4-1:

$$I_{SP_{1 \text{ second}}} = I_{SP_{SS}} - \frac{2500}{1000} = I_{SP} - 2.5$$

For 1-second pulse

$I_{SP} \leq 2.5$  seconds less than steady state

$I_{SP} \leq 0.83$  percent less than steady state

### Syncom Estimated Overall Mission $I_{SP}$

#### 1. Test Results for X19150 Engine:

$I_{SP}$  - continuous: 301.5 seconds - 5-pound thrust  
 operation: 285 seconds - 3-pound thrust

#### 2. Predicted for 1-Second Pulse:

300 seconds - 5 pounds  
 283 seconds - 3 pounds

\*15 percent more propellant is wasted for the Apollo DV compared to the Syncom DV at minimum thrust level.

3. Predicted for 0.1-Second Pulse:

$$301.5 - 15 = 286.5 \text{ seconds} - 5 \text{ pounds}$$

$$285 - 22 = 263 \text{ seconds} - 3 \text{ pounds}$$

$$(I_{SP_{SS}} - I_{SP_{0.1 \text{ second}}}) \cong (I_{SP_{SS}} - I_{SP_{0.1 \text{ second}}})_{\text{Apollo}}$$

$$\times \frac{\text{percent wasted propellant (Syncom)}}{\text{percent wasted propellant (Apollo)}}$$

$$(I_{SP_{SS}} - I_{SP_{0.1 \text{ second}}}) 5 \text{ pounds} \cong 25 \times \frac{1}{1.73} = 15 \text{ seconds}$$

$$(I_{SP_{SS}} - I_{SP_{0.1 \text{ second}}}) 3 \text{ pounds} \cong 25 \times \frac{1}{1.15} = 22 \text{ seconds}$$

4. Effective  $I_{SP}$  for Required  $I_T$ 

$$I_T \propto F$$

$$I_{SP_{EFF}} \text{ for 1-second pulse} = \frac{5 \times 300 + 3 \times 283}{8} = 294 \text{ seconds}$$

$$I_{SP_{EFF}} \text{ for 0.1 second pulse} = \frac{5 \times 286.5 + 3 \times 263}{8} = 278 \text{ seconds}$$

5. Estimated Overall Mission  $I_{SP}$ 

80 percent 1 second pulses

20 percent 0.1 second pulses

$$I_{SP_{MISSION}} = 0.80 \times 294 + 0.20 \times 278 = 292 \text{ seconds}$$

|  |
|--|
| $I_{SP_{MISSION}} = 292 \text{ seconds}$ |
|--|

## TELEMETRY AND COMMAND ANTENNA DESIGN

The antenna configuration will be determined by pattern tests conducted on a spacecraft mockup recently completed (Figure 4-28). The proposed telemetry and command system contains two sets of antennas, four diplexers, and two hybrid baluns to interconnect the four telemetry transmitters and four command receivers.

### Telemetry Encoder

At the time of publication of this report, the telemetry system had not been completely defined. The alternatives being considered by Goddard Space Flight Center and Hughes are the Goddard standard PFM system (wide-band) and the Syncom I type telemetry system (narrow-band). The following parameters have been determined for each of the two systems:

|                             | <u>Narrow-band</u> | <u>Wide-band</u> |
|-----------------------------|--------------------|------------------|
| Subcarrier center frequency | 14.5 kcps          | 10 kcps          |
| Subcarrier deviation        | $\pm 7.5$ percent  | $\pm 50$ percent |
| Channel rate                | 3.03 second        | 48.5 second      |
| Number of channels          | 64                 | 128              |

Effort has begun on encoder circuits which will be the same for either system; presently under development are the commutator, dc-dc converter, and solar sensor amplifier.

### Command Decoder

The preliminary design of the dual-mode command decoder was completed, and two additional single-mode systems were designed to serve as alternate means of decoder implementation. A design inventory was prepared and a preliminary design review of the decoder was held on 17 May 1963.

No errors were found in the preliminary design of the decoder at the design review, and following the review the dual-mode system was recommended for use. An analysis concerning the signal levels necessary for decoder operations was made and it was found that there would be no significant problem in the implementation of the dual-mode system provided that the separation between the message tones is increased.

Representatives of NASA briefly reviewed the decoder on 22 May 1963 and indicated that the separation between the tones would be increased.

### Command Regulator (212 Unit)

This regulator must be capable of supplying 50 ma at  $\pm 24$  volts. Unlike all other regulators which act as on-off switches for their respective loads, this regulator must be self-starting. Whenever the unregulated bus

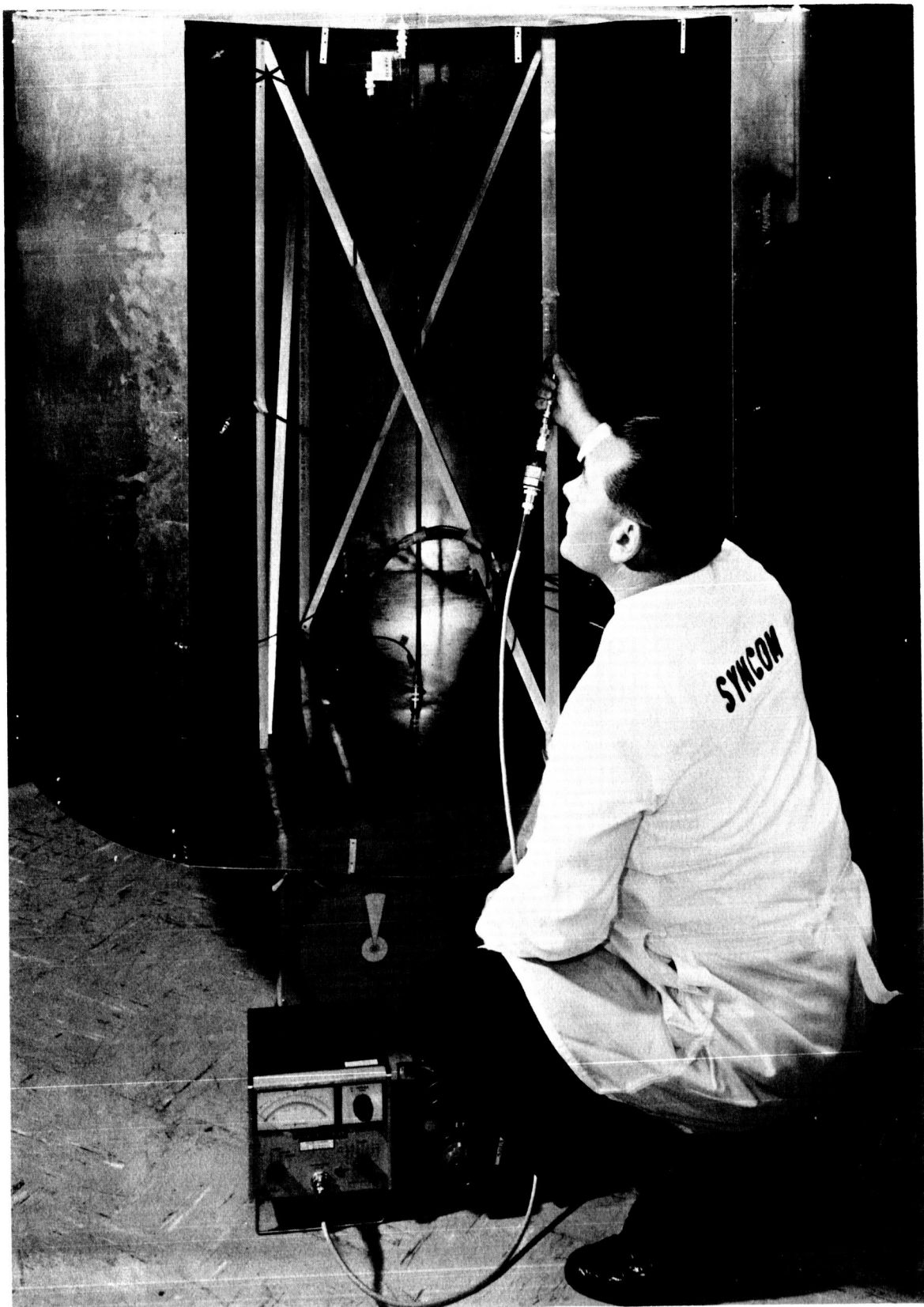


Figure 4-28. Telemetry and Command  
Antenna System Test Mockup

exceeds (becomes more negative than) -17 volts, the regulator will automatically turn on, but with low output voltage. When the input exceeds -26 volts, the regulator will supply  $\pm 24$  volts  $\pm 1$  percent. It is desirable that the +24- and -24-volt outputs track each other. For this reason, the -24-volt output will be taken from the same dc-dc converter secondary as the +24-volt output. The series regulator -24-volt sensing will be taken from the converter -24-volt output for better load regulation. This method of remote sensing is identical to that discussed in more detail relative to the PACE Regulator (160 Unit).

#### Telemetry Regulator (222 Unit)

This regulator must supply 350 ma at -24 volts. It will be a Syncom I type series regulator, with a relatively sharp overcurrent trip circuit added. Except for the level at which the overcurrent trip is set, this regulator will be identical to the transponder receiver F. T. mode regulator (102 Unit).

### ELECTRICAL POWER

As a result of the NASA design review comments, analysis is proceeding on the electrical power system configuration. Investigation is continuing on the impact to the electrical power system of: 1) 3-ohm centimeter solar cells in lieu of 1-ohm centimeter, and 2) 64 cells in series instead of 29 cells.

Computation on radiation damage will be pursued as soon as updated environmental information is available from Goddard Space Flight Center.

#### Battery Charge Regulator (251 Unit)

The regulator design previously reported is being reevaluated with the objective of achieving greater simplicity. The results should be available during the next report period.

### STRUCTURE

#### Structural Design

The structural and equipment arrangements of the spacecraft have been revised to attain new objectives. Included, but not limited to, are increased space for quadrant electronic packages, improved wiring harness installation, increased thermal conductivity of structural elements, accommodation of the revised apogee motor and the revised control jet system, and improved maintenance and installation access to all components.

The primary structure for the spacecraft is subdivided into three sections: an aft thrust tube, a central thrust tube, and a forward equipment

support structure. The aft and central thrust tubes are fabricated as a single assembly. The forward equipment support section is to be made readily removable to facilitate fabrication, assembly, and maintenance of the vehicle. The general arrangement is shown in Figure 4-29 and an exploded view of the spacecraft structure and control system installation is shown in Figure 4-30. An illustration of the spacecraft is shown in Figure 4-31.

The aft face of the aft thrust tube attaches to the Agena adapter section by a V-band clamp. Radial ribs of aluminum alloy are attached to the aft structure to support the fuel tanks and the aft electronic components and to provide thermal paths for heat distribution. The brackets for the aft batteries are attached to these ribs, as are the aft hoist support fittings. The antenna package is supported inside the thrust tube by a structure similar to that previously used except that it has been reduced in complexity and weight. A ring frame is used at each end of the aft thrust tube to distribute lateral loads into the thrust tube and to provide balancing reactions for longitudinal loads in the ribs. The upper ring frame, adjacent to the apogee motor attach points, is of a high degree of stiffness to avoid introduction of loads into the apogee motor case.

The central structure is a thrust tube of aluminum alloy sheet with external longitudinal stiffeners. The diameter of this tube provides clearance about the apogee motor for a motor installation protective sleeve and structure thermal insulation. The quadrant electronic packages, the fuel tank forward supports, and the support for the forward batteries are attached to the external longitudinal stiffeners. The forward ends of the stiffeners provide for attachment of the removable forward equipment support structure and the forward hoist fittings.

The forward equipment support structure is a box section ring frame to serve as a support for the forward mounted components. The box section will provide rigid support in all directions to the equipment components. A tubular truss structure connects the ring frame to the central structure attach points. Although the truss structure is efficient, studies are being made to optimize this section. These investigations will include beaded sheet metal and Vierendeel truss arrangements. The complete forward equipment support structure assembly is removable to provide for the installation of the jets and fuel system assembly.

The three attachment points for each solar cell support panel are on 120-degree spaced radial lines from the center of the panel and are located near the edge of the panel. These attach points provide constraint normal to the plane of the panel, and, within the plane of the panel they provide constraint only in the direction perpendicular to the radial lines from the center. The attachments are hinged or slotted so that no reaction loads will be applied in a direction of the radial lines. This arrangement ensures that deformation of the spacecraft structure will not introduce loads into the solar cell support panels. This arrangement is shown in Figure 4-32.

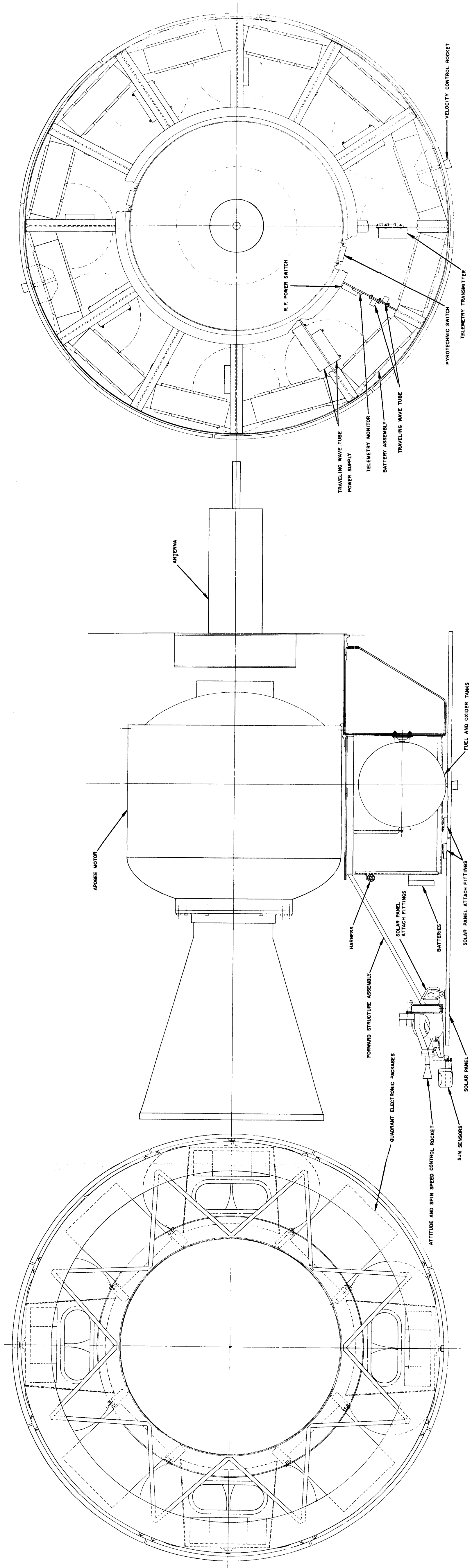


Figure 4-29. General Arrangement Syncom II

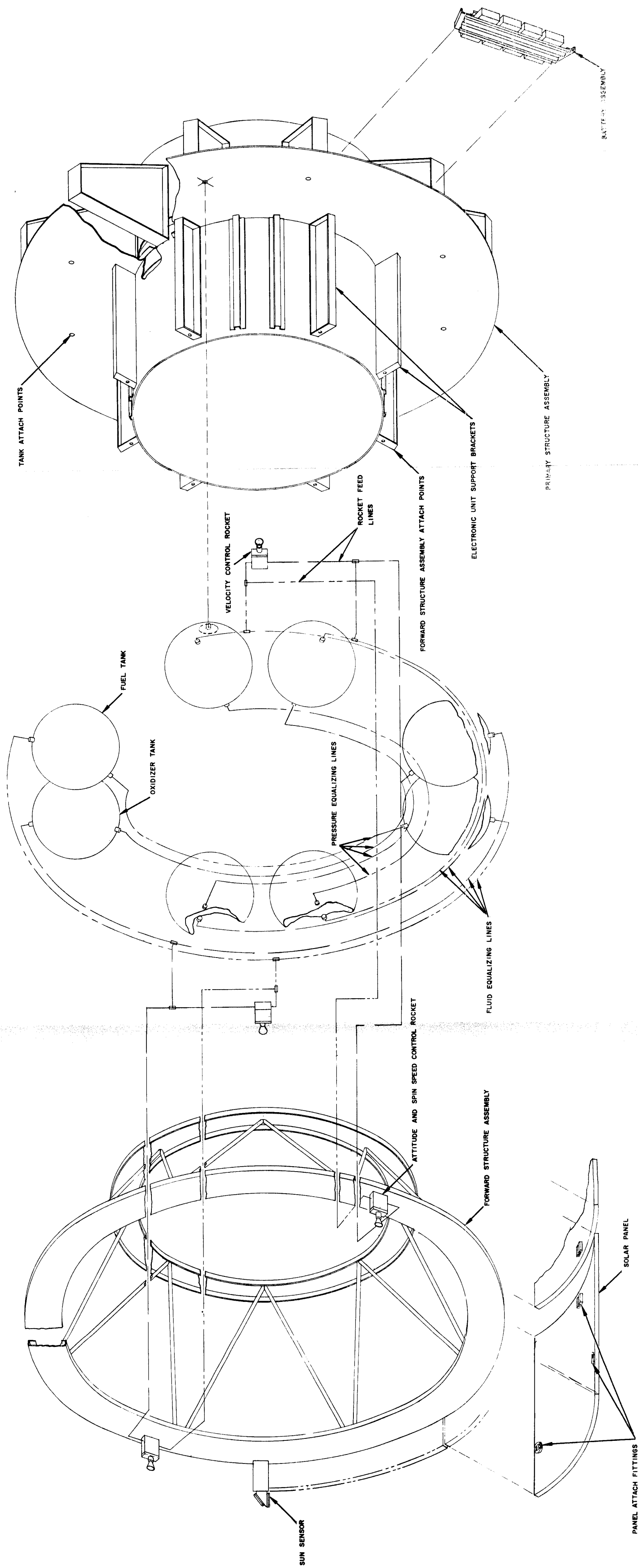


Figure 4-30. Exploded View - Syncom II



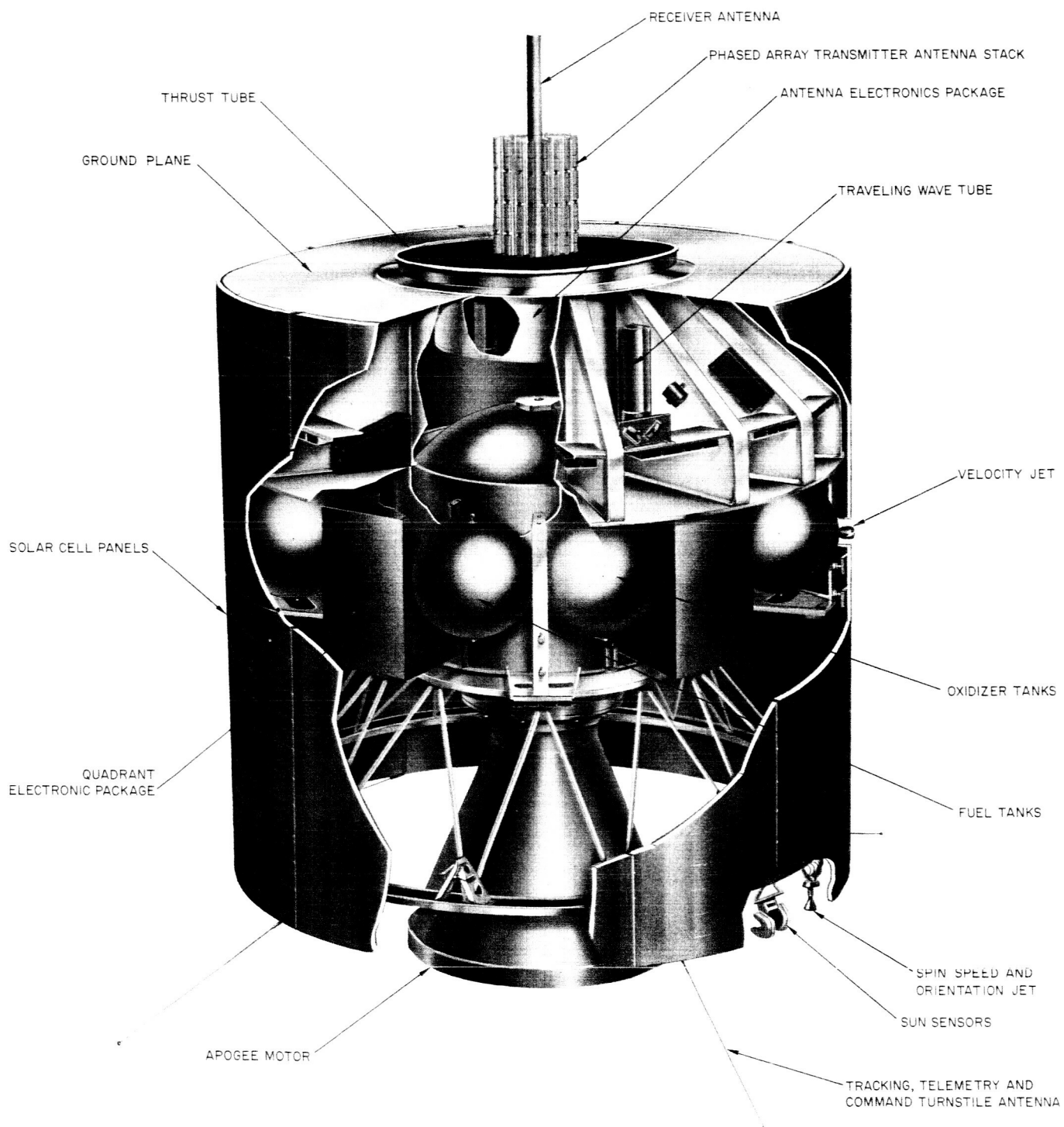


Figure 4-31. Mark II Interior

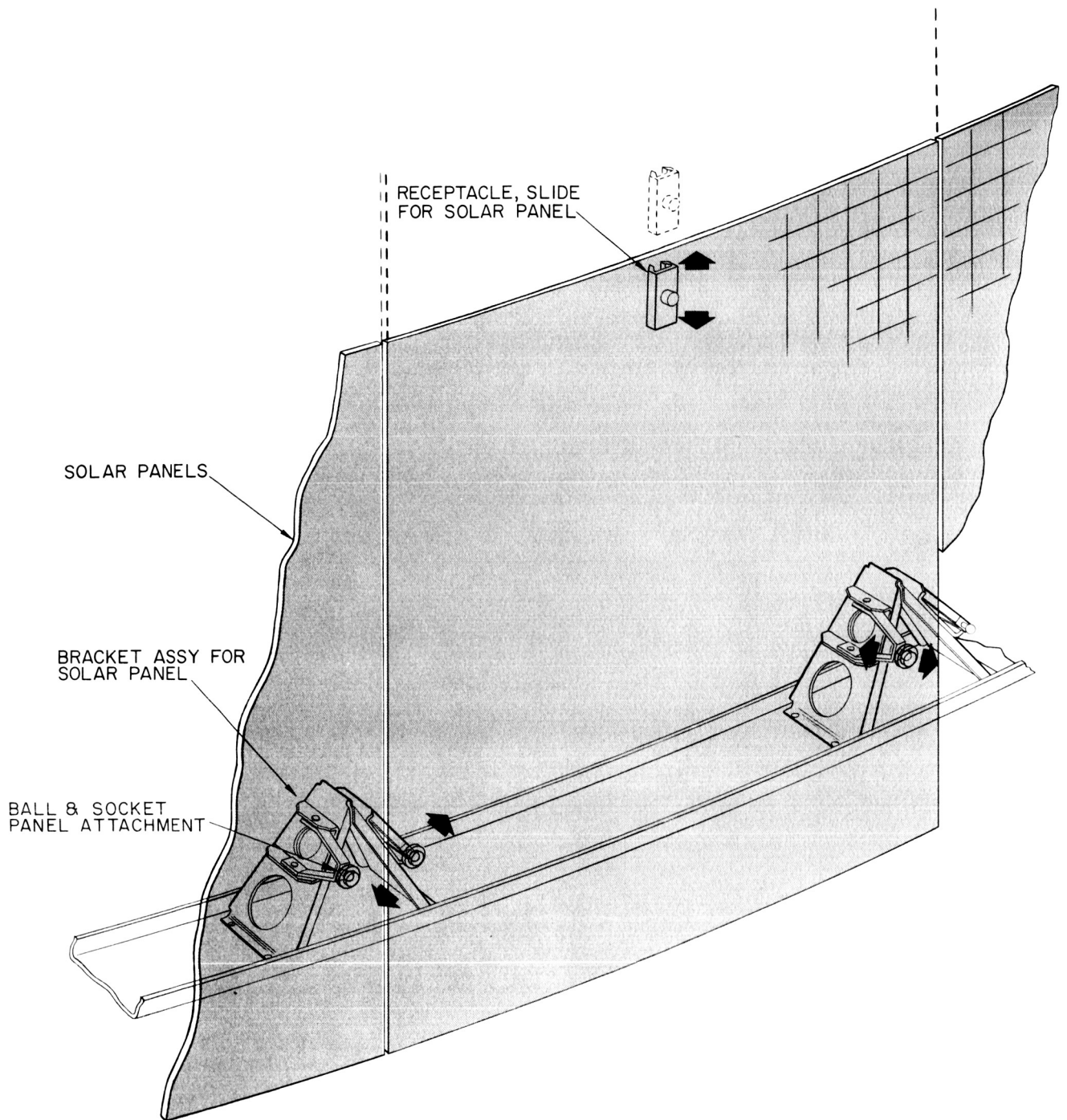


Figure 4-32. Solar Cells Support Panels Attachment

## Structural Analysis

Stress effort during this report period has been concentrated in the following areas:

- 1) Analysis and redesign of the structure supporting the spin rate control engine, sun sensors, and forward solar panels.
- 2) Analysis and redesign of the thrust tube to determine the optimum section for strength, stiffness, and mating with the JPL apogee motor.

Optimization studies for the thrust tube include a comparison of the EI, AE, AG, and deflection curves for the T-1 structure and proposed T-2 configurations. These curves for the T-1 structure are presented in Figures 4-33 and 4-34 and will be delivered to Lockheed/Sunnyvale for inclusion in their dynamic studies.

A new axial vibration model has been devised to reflect design changes in the T-2 engineering model and also to permit a larger number of degrees of freedom than was allowed in the dynamics model for T-1. The model contains a total of 65 masses and approximately 50 springs. Because of symmetry and allowing for constraints, this model will permit approximately 15 degrees of freedom. At present, the equations of motion are being developed for this model.

### Mass Properties Analysis (Weight and Balance)

Table 4-3 summarizes the mass property data in the planned launch configuration. The current statement includes JPL estimates of weight and moment-of-inertia data for the apogee motor and Marquardt estimates of weight data for the reaction control system. Prelaunch and burnout weight, balance and moment-of-inertia data for the JPL apogee motor as calculated by Hughes are not included at this time. A reduction of 30 pounds and nominal changes in balance and inertia data are indicated, subject to confirmation by JPL.

Major structural design changes and a comprehensive weight analysis of the electronics subsystem are in progress. The anticipated weight changes are not included in this report; therefore, significant revisions of this weight statement are forthcoming.

Weight changes since the last report are as follows:

| <u>Description of Change</u>   | <u>Weight, pounds</u> |
|--|-----------------------|
| Solar panel - increased length and thickness, mounting provisions changed, and retainers eliminated. | + 5.5                 |

| <u>Description of Change</u>  | <u>Weight, pounds</u> |
|---|-----------------------|
| Dry reaction control system - growth allowance eliminated and purge fitting added per Marquardt estimate. | -0.6                  |
| Ballast - reduced to compensate for space-craft growth.   | -4.9                  |

TABLE 4-3. ESTIMATED WEIGHT STATUS  
Planned Launch Configuration

| Subsystem   | Weight* | Weight, pounds | $\phi^{**}$ | $\phi^{***}$                     |                                  |      |
|---|---------|----------------|-------------|----------------------------------|----------------------------------|------|
| Electronics   |         | 134.7          | 0.215       | 0.089                            |                                  |      |
| Wire harness  |         | 19.9           | 0.032       | 0.013                            |                                  |      |
| Power supply  | +5.5    | 113.5          | 0.181       | 0.075                            |                                  |      |
| Control   | -0.6    | 48.9           | 0.078       | 0.032                            |                                  |      |
| Propulsion  |         | 122.2          | 0.195       | 0.081                            |                                  |      |
| Structure   |         | 138.3          | 0.221       | 0.091                            |                                  |      |
| Miscellaneous                                       |         | 19.9           | 0.032       | 0.013                            |                                  |      |
| Ballast   | -4.9    | 28.1           | 0.045       | 0.019                            |                                  |      |
|   |         |                |             |                                  |                                  |      |
|   |         | Weight, pounds | Z-Z, inches | $I_{Z-z}$ , slug-ft <sup>2</sup> | $I_{X-x}$ , slug-ft <sup>2</sup> | R/P  |
| Final orbit condition                               |         | (625.4)        | 23.6        | 56.7                             | 48.2                             | 1.18 |
| N <sub>2</sub> pressurization                       |         | 2.9            |             |                                  |                                  |      |
| N <sub>2</sub> H <sub>3</sub> -CH <sub>3</sub> fuel |         | 53.1           |             |                                  |                                  |      |
| N <sub>2</sub> O <sub>4</sub> oxidizer              |         | 84.3           |             |                                  |                                  |      |
| Total at apogee burnout                             |         | (765.7)        | 23.6        | 71.0                             | 55.3                             | 1.28 |
| Apogee motor propellant                             |         | 752.3          |             |                                  |                                  |      |
| Total payload at separation                         |         | (1518.0)       | 24.6        | 87.8                             | 72.5                             | 1.21 |

\*Change in subsystem weight since last report.

\*\*Ratio of subsystem weight to final orbit condition weight.

\*\*\*Ratio of subsystem weight to total payload at separation.

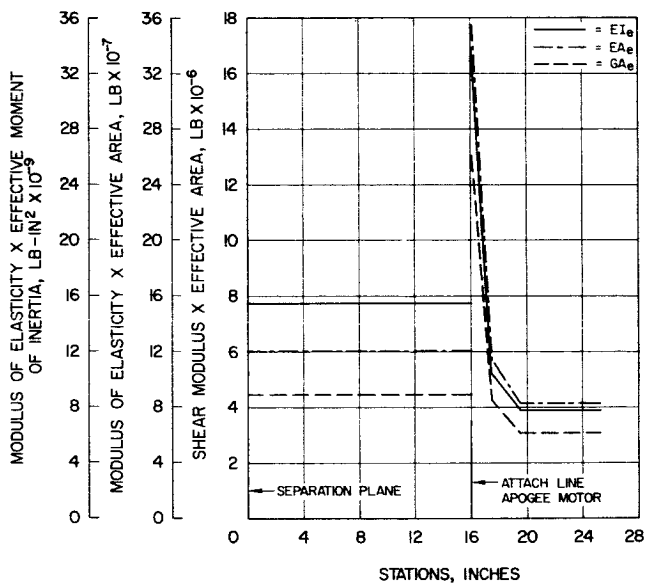


Figure 4-33. Stiffness Distribution

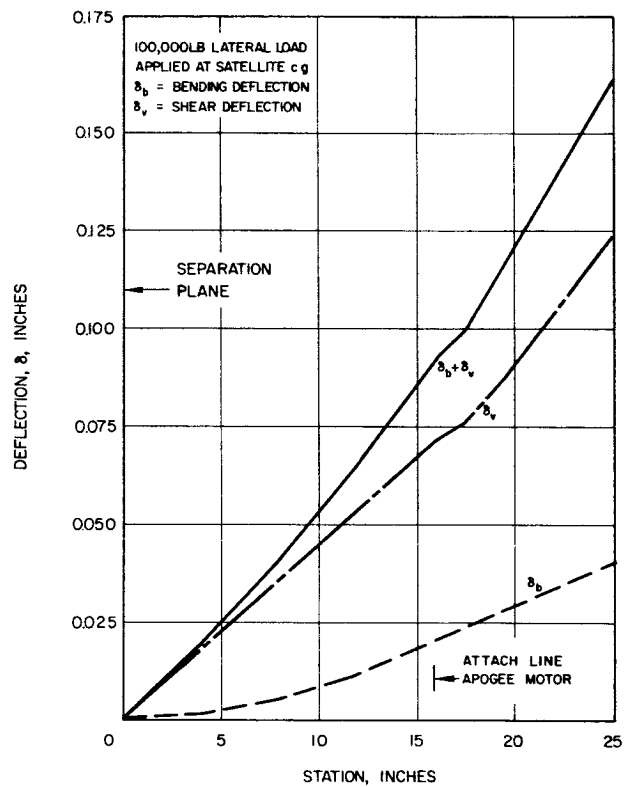


Figure 4-34. Deflection versus Station

### Dynamic Test Status and Preliminary Data (T-1)

Vibration tests Nos. 3 through 9 and the shaker investigations were completed this month (Figure 4-35). The test schedule is given in Table 4-4. Test 10 will be completed early in June.

TABLE 4-4. VIBRATION TEST PROGRAM

| Test | Input Location        | Excitation Axis | Sinusoidal Input Level |
|------|-----------------------|-----------------|------------------------|
| 1    | Agena interface       | Longitudinal    | 1/2 to 1 g peak        |
| 2    | Agena interface       | Longitudinal    | Qualification          |
| -    | Shaker investigations |                 |                        |
| 3    | Agena interface       | Lateral -1      | 1/2 to 1 g             |
| 4    | Agena interface       | Lateral -1      | Qualification          |
| 5    | Agena interface       | Lateral -2      | Qualification          |
| 6    | Apogee motor attach   | Lateral -1      | 1/2 to 1 g peak        |
| 7    | Apogee motor attach   | Lateral -1      | Qualification          |
| 8    | Apogee motor attach   | Lateral -2      | Qualification          |
| 9    | Apogee motor attach   | Longitudinal    | 1/2 to 1 g             |
| 10   | Apogee motor attach   | Longitudinal    | Qualification          |

Acceleration response data from these tests are being analyzed and will be reported when available.

The measured fundamental frequencies of the spacecraft are given in Table 4-5. These frequencies have not been corrected for effects of the shaker armature mass or suspension spring rates. The corrected frequencies are expected to be slightly lower than the measured values.

TABLE 4-5. MEASURED FREQUENCIES

| Direction    | Attach Point    | Frequency |
|--------------|-----------------|-----------|
| Lateral      | Agena interface | 49        |
| Longitudinal | Agena interface | 120       |

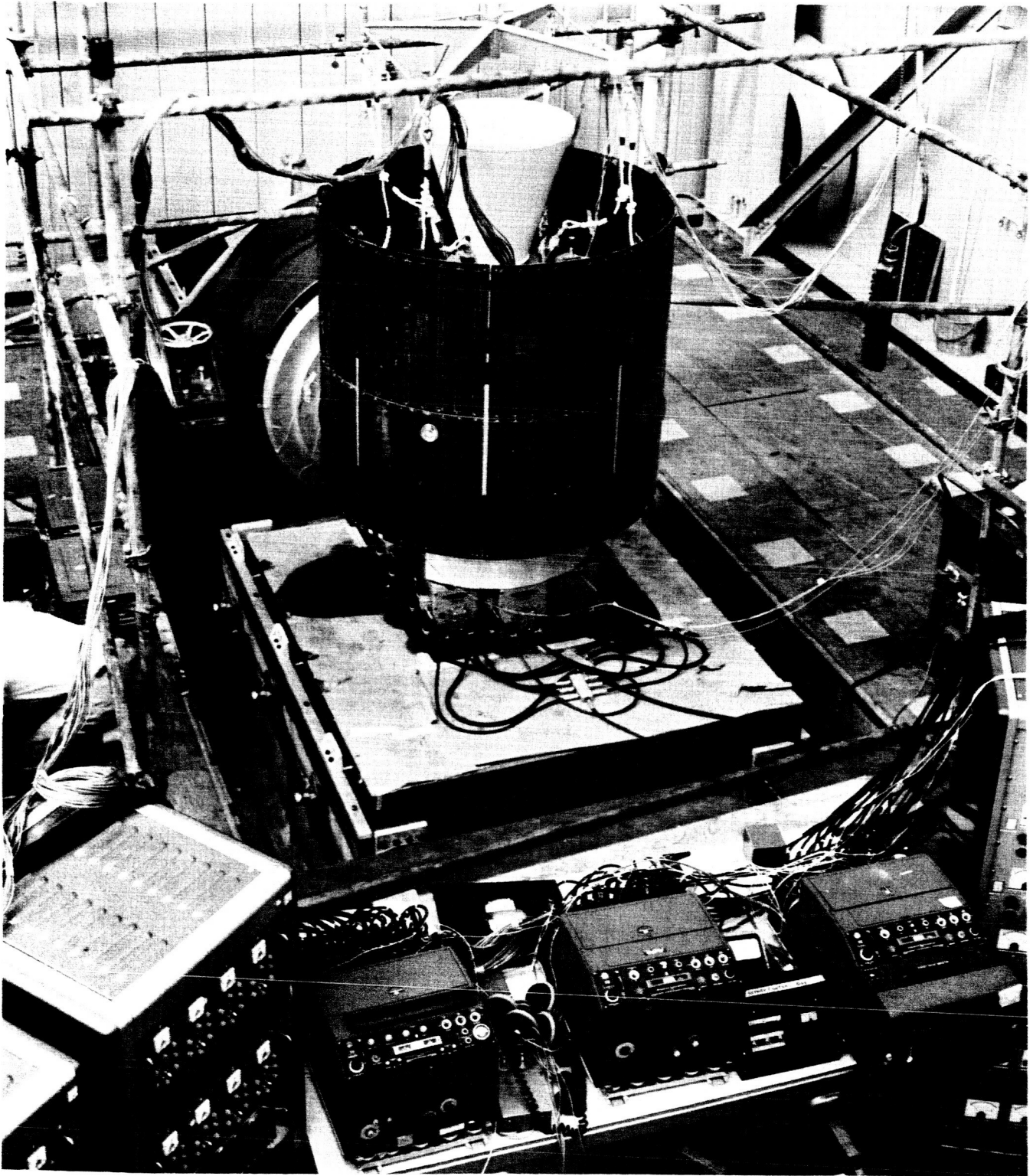


Figure 4-35. Syncom II Spacecraft  
Undergoing Vibration Testing

The following structural damage has occurred in tests 3 through 8:

- 3: Velocity rocket motor truss failed (discussed below).
- 4: None.
- 5: Small face sheet separations at the corners of several solar panels.
- 6: Velocity rocket motor truss failed.
- 7: Sun sensor bracket cracked, velocity rocket motor truss failed.
- 8: Small face sheet separations at the corners of several solar panels.
- 9: None.

The failures of the velocity rocket motor truss were similar to that shown in Figure 4-36 which shows the failure of test 3. All of the failures occurred in the vicinity of the rocket motor; in each case the tubing was cracked or broken near a weld. The failures are being examined and will be discussed in the vibration test report.

## SPACECRAFT THERMAL CONTROL

Thermal analysis of the spacecraft is continuing with parametric studies of a one-fourth segment of the spacecraft. Since the design of the spacecraft imposes symmetry among the four quadrants, the thermal nodal analysis has been prepared to thermally simulate a single quadrant. The concept has resulted in a nodal network comprised of approximately 60 to 80 nodes as compared to the 250-node representation of Syncom I. Parameters that are being varied in the analysis are:

- 1) Internal power dissipation
- 2) Solar angle of incidence on the spacecraft
- 3) Variance of surface thermal properties such as solar cells, white paint, polished aluminum, and vapor-deposited aluminum.
- 4) Surface thermal property degradation due to aging and ultraviolet deterioration.

In addition, operational effects on spacecraft temperatures are to be investigated utilizing the existing nodal model. These are:

- 1) Eclipse
- 2) Reaction jet firing
- 3) Apogee motor firing



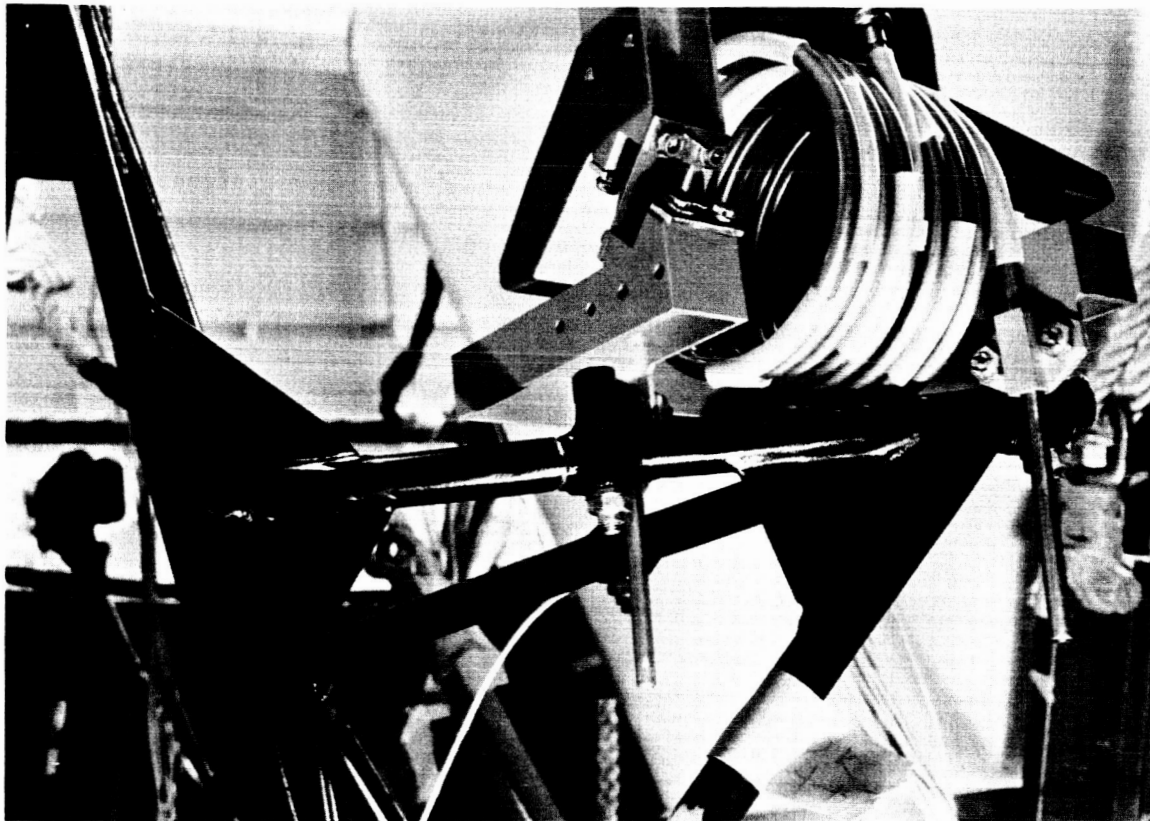


Figure 4-36. Velocity Rocket Truss  
Failure

A relevant factor in determining thermal performance of the vehicle is the long-term life requirement of the spacecraft thermal surface materials. Efforts are under way to coordinate investigation of aging effects in an ultraviolet environment on potential outside surface materials. The outcome of this investigation will determine the need or lack of need for an active thermal control system to smooth out the possible perturbations that may occur in spacecraft temperatures.

Effort is continuing in the investigation of the specular absorptance of the solar cells. Since the cells are absorbing solar energy at all angles because of the geometry of the spacecraft, the angular dependence of solar absorptance must be known for proper energy balance on the spacecraft.

To assist in the parametric study of spacecraft temperatures, a simplified single-node, multisurface, analytical model has been prepared. Parametric investigations using this model will yield preliminary bulk average vehicle temperatures. The results can be used to verify computer results from the multinode network. It is recognized, however, that the simplified model cannot yield information as to internal thermal gradients, which by the nature of the spacecraft geometry are expected to be small.

#### APOGEE INJECTION ROCKET MOTOR

The apogee motor, under development by JPL, is a scale-up of the Syncom I Starfinder apogee motor. The initial design effort has been completed except for the flight nozzle. Seven flight-weight cases have been ordered and four heavy-wall cases have been received. Fabrication tooling has been received and the first heavy-wall motor is scheduled for loading with inert propellant in June 1963. Two of the scheduled four subscale motor tests, using Syncom I hardware, have been successful.

The first test of a heavy-wall motor with a truncated nozzle is scheduled for late June 1963. Additional igniter tests will be conducted using an ignition test motor and the ignition characteristics will be evaluated to determine performance of a pyrotechnic-type igniter and a controlled-pressure igniter.

## 5. SPACECRAFT RELIABILITY AND QUALITY ASSURANCE

### RELIABILITY ANALYSIS

#### Power Supply Analysis

A meeting between the cognizant power supply engineering group and systems reliability was held to clarify the solar array reliability analysis.

The reliability analysis presented in the Advanced Syncom Summary Report, 31 March 1963, pages 7 through 11, presents a solar array analysis based on the probability of survival for at least one string per array string group. Therefore, the results represent a probability of 0.99 that at least one-third of the power output be available at the end of the 5-year mission time. This is sufficient power for one quadrant operation. It is apparent that this analysis should now be extended to determine the probable power degradation versus orbital time. The power loss function can be generated in terms of the number of cell failures per string group. The effect of random cell failures upon power output of a string group is a deterministic function similar to that for a single cell failure within a string group (Figure 5-1).

A reliability analysis of the solar array will be performed during this phase of the program incorporating this approach. It is intended that such a solar array analysis be applied to both a single-supply approach and the dual-supply systems.

#### Failure Mode and Effects Analysis

During this period major effort was expended in reviewing, revising, and extending failure mode and effects analysis procedures. This technique is of prime importance in focusing the attention of design engineers on reliability. It is particularly useful in providing thorough documentation of design techniques affecting reliability.

### QUALITY ASSURANCE

Special Quality Control Instructions (QCIs) are being prepared. To date the following preliminary QCIs have been written to control procurement and fabrication documents:

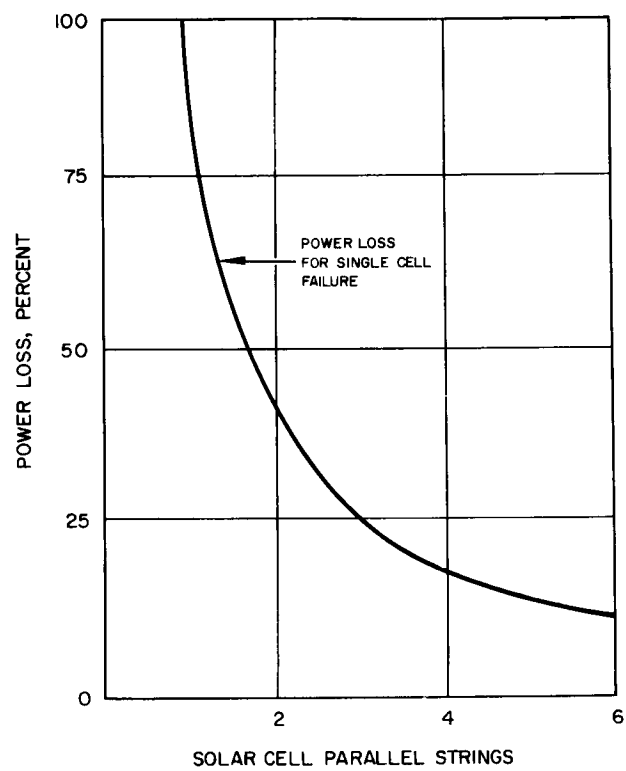


Figure 5-1. Power Loss versus Parallel Strings

- QCI 3.13    Syncom II – Screening Purchase Orders for Subcontract Items
- QCI 3.13    Syncom II – Screening Manufacturing Requests for Quality Control Requirements
- QCI 3.13    Syncom II – Screening Work Orders for Quality Control Requirements
- QCI 3.13    Syncom II – Screening Assist Work Authorizations

In-process inspection is being performed on items for the engineering models being fabricated during this phase of the contract.

An inspection planning group is being established to develop inspection instructions for each spacecraft control item and for critical assemblies. The instructions will be based on released drawings and shop work orders. An inspection check list will be included as part of each instruction. The degree of completion of this task by 30 August 1963 will depend on the availability of control item drawings.

The supplier control group in conjunction with project reliability and engineering personnel has prepared the following:

Program plan for the supplier control group

Manpower utilization chart

Training program for the source liaison engineers

The supplier control group has prepared a quality plan for Marquardt, the subcontractor responsible for the bipropellant reaction control system. This quality plan has been coordinated with project personnel to ensure the inclusion of all contractual requirements.

A survey was made of the supplier's facility to establish his quality effort and knowledge in regard to this contract. It was evident that the supplier's quality control and reliability organization required some buildup to be ready for the manufacture of the quality assurance model. As a result of this survey, a meeting was held with Marquardt's reliability, quality control, and engineering personnel and similar Hughes personnel to discuss a quality assurance program. A resurvey will be conducted at a later date to determine the extent of the supplier's effort.

The supplier control group has conducted facility surveys for conformance to NPC 200-3 at 15 potential semiconductor suppliers and obtained past performance data to establish their ability to provide high-reliability semiconductors. As a result of these surveys, additional work will be performed in liaison with these vendors to ensure their performance is in accordance with high reliability aims. Quality plans are presently being established for all vendors.

Similarly, six passive device suppliers have been surveyed and evaluated. In addition, approximately ten more surveys are to be performed. Liaison will be established with all these suppliers as is necessary to the program. Quality plans are presently being established for all vendors. One vendor, Rantec Corporation, is considered to require a long lead time for the procurement of hybrid filters. The supplier control group has conducted an initial survey of the vendor's facility and is performing active surveillance at this time.

#### Survey of Electronic Parts Manufacturers

Survey teams with members representing Hughes project management, quality control, component engineering, and purchasing have visited many companies whose products will be used. The companies were selected on the basis of advanced bills of materials, derived from engineering drawings in a preliminary status. A reliability program was discussed with each company's personnel and the companies were assessed on their experience and apparent ability to comply with Hughes requirements. Manufacturers visited were as follows:

| <u>Manufacturer</u>            | <u>Location</u>                    | <u>Date</u> |
|--------------------------------|------------------------------------|-------------|
| Hoffman                        | El Monte, California               | 15 April    |
| Pacific Semiconductor          | Hawthorne, California              | 17 April    |
| Hughes                         | Newport, California                | 18 April    |
| Motorola                       | Phoenix, Arizona                   | 22 April    |
| Texas Instruments              | Dallas, Texas                      | 23 April    |
| Philco                         | Lansdale, Pennsylvania             | 25 April    |
| Silicon Transistor Corporation | Carle Place, Long Island, New York | 26 April    |
| Microwave Assoc.               | Burlington, Massachusetts          | 22 April    |
| Sylvania                       | Woburn, Massachusetts              | 23 April    |
| General Electric               | Syracuse, New York                 | 24 April    |
| Fairchild Transistor           | Mountain View, California          | 6 May       |
| Fairchild Diode                | San Rafael, California             | 7 May       |
| Raytheon                       | Mountain View, California          | 8 May       |
| Amelco                         | Mountain View, California          | 9 May       |
| Kemet                          | Cleveland, Ohio                    | 16 May      |
| Allen-Bradley                  | Milwaukee, Wisconsin               | 15 May      |
| Erie Resistor                  | Erie, Pennsylvania                 | 17 May      |

| <u>Manufacturer</u> | <u>Location</u>            | <u>Date</u> |
|---------------------|----------------------------|-------------|
| Vitramon            | Bridgeport, Connecticut    | 18 May      |
| Sprague Electric    | North Adams, Massachusetts | 19 May      |
| Corning Glass       | Raleigh, North Carolina    | 20 May      |
| Texas Instruments   | Dallas, Texas              | 21 May      |

Manufacturers to be visited after June 10 are:

|          |                            |
|----------|----------------------------|
| Johanson | Boonton, New Jersey        |
| Maida    | Hampton, Virginia          |
| IRC      | Philadelphia, Pennsylvania |
| Elmenco  | Willamantic, Connecticut   |
| Microdot | Los Angeles, California    |
| Cannon   | Los Angeles, California    |
| Daven    | Livingston, New Jersey     |

The parts reliability program consists of several phases which are covered in a specification addendum for approved parts: manufacture, test, and power age. The companies were given a standard quality control survey in compliance with NASA Quality Document NPC 200-3, failure reporting and corrective action, and a quality control checklist applied uniformly by Hughes to any manufacturer of parts ordered for any program. Manufacturing processes were observed for cleanliness, observance of rules, inconsistencies, and the general appearance of the plant and personnel. In-process inspection techniques were checked for adequacy; the parts program calls for special visual and X-ray inspection if required to supplement a marginal area in manufacture. Electrical test data acquisition and handling plans are being prepared, necessitated by the large quantity of information expected from the power aging program. Methods of data presentation and reduction were discussed, and several semiconductor companies are able to provide analysis for use in selecting the parts most suitable for flight electronics.

One important result of the survey was the understanding gained of manufacturers' experience in improving reliability. A large dollar saving will be achieved by making maximum use of tests performed in the manufacturing process and by not specifying costly redundant tests. An additional gain will be made by ordering parts from lots already qualified to Minuteman or Polaris specifications, saving expensive lot qualification tests.

Preliminary cost estimates are now being received from the manufacturers. Effort is continuing to update the bill of materials which will allow a reasonably accurate cost estimate to be available in July. An important byproduct of reviewing the bill of materials has been the reduction of semiconductor types from 65 to approximately 40 with further reduction expected.

## 6. MATERIAL, PROCESSES, AND COMPONENTS

### CRITICAL COMPONENT TEST PLANS

It is the purpose of these test plans to evaluate and demonstrate that the applicable units will perform adequately during the Syncom II launch and orbital phases.

#### Ferrite Switches

There will be eight ferrite switches used in the Syncom II spacecraft. Four will operate at 6 gc and four at 4 gc. One 6-gc switch is located in each of the four transponders at the input of the two receivers (frequency translation and multiple access) to completely separate them rather than depending on a back-biased crystal to control signal flow. One 4-gc switch is used in each of the four transponders to connect either of two redundant traveling-wave tubes to the transmitting antenna.

The switch is a three-part circulator where the direction of circulation is controlled by the direction of a permanent magnetic field. A current is applied to an external coil which sets up the magnetic field inside the circulator. Once the direction of the magnetic field has been changed the external current is removed.

The testing of the ferrite switch is planned for accomplishment in two phases. The first phase will be preliminary tests conducted during the present contractual period to demonstrate the ability of the switch to withstand the expected qualification environments. The second phase will be conducted on switches that have been fabricated to final released drawing e.g., flight-quality hardware.

#### Phase I

There will be a total of four each of the 4-gc and 6-gc switches fabricated. Three each of these switches will be subjected to the expected environmental conditions. Performance test measurements will consist of, but not be limited to, the following: insertion loss, drive power, cross-coupling between two input ports, and input voltage standing wave ratio. One each of switches will be subjected to the qualification level environments.



## Phase II

A total of eight ferrite switches will be tested. Four of these will operate in the 4-gc range and four at the 6-gc range. They will be divided into four groups. A group will consist of one 4-gc and one 6-gc switch. They will be subjected to the following environments in the order shown below.

- 1) Vibration and shock
- 2) Acceleration (spin and boost)
- 3) Acoustical noise
- 4) Thermal-vacuum

| <u>Group</u> | <u>Environment Sequence</u> |
|--------------|-----------------------------|
| 1            | 1, 2, 3, 4                  |
| 2            | 2, 3, 4, 1                  |
| 3            | 3, 4, 1, 2                  |
| 4            | 4, 1, 2, 3                  |

After successful completion of the above, each switch shall be electrically cycled 1000 times. RF measurements will be made at the conclusion of each 100 cycles.

## Phase Shifter

There are eight phase shifters used in the spacecraft, each consisting of the following:

- 1) Input couplers — couple the outputs of the power splitter to the ferrite sections.
- 2) Ferrite sections — consist of a tube of ferrite in a circular waveguide inside a four-pole, two-phase electromagnetic field coil.
- 3) Output coupler — converts the RF output of the ferrite sections into two equal amplitude signals with opposite polarity phase shifts to drive the antenna elements.

The ferrite sections are also driven by the electromagnetic field supplied by signals from the phased-array control electronics (PACE) which provide the correct phasing angles so that the output of all antenna elements are properly phased to form a beam directed toward the earth.

The testing is planned for accomplishment in two phases. The first phase will consist of the following:

Three phase shifters will be tested during this contractual period. The testing will be conducted under temperature conditions and thermal-vacuum conditions. Temperature tests at laboratory ambient pressure will be performed at 0 and at 150° F. These two temperatures are well below and above those expected during orbital operation. Phase shift between the output and input will be measured as a function of current through the motor stator winding. RF losses and power output of the two outputs will be measured and compared. The phase shifters will then be tested in a thermal-vacuum chamber. The vacuum will be  $10^{-5}$  Torr or less. The phase shifters in the vacuum will be stabilized at a temperature of 40° F and remain at this condition for a period of 48 hours. The same performance tests as stated above will be conducted once every 24 hours. While remaining in the vacuum the temperature of the phase shifters will be increased to 130° F and remain at this condition for a period of 48 hours. Performance tests will be conducted once every 24 hours.

The phase shifters will then be subjected to the qualification vibration environment. Performance tests will be conducted before and after the vibration tests.

The second phase of the test program will be accomplished at the time the final communications antennas are available. This test plan will consist of testing one communications antenna (Major Control Item) which comprises the following units: eight RF phase shifters, one phase shifter power splitter, one antenna array output coupler, and one transmitter antenna array.

Performance measurements will be conducted in accordance with the applicable test specifications for each of the above units. The communications antenna will be subjected to the following qualification level environments:

- 1) Vibration.
- 2) Shock.
- 3) Acceleration (spin and boost).
- 4) Thermal-vacuum.

#### Traveling-Wave Tubes

During the period covered by the present contract, a total of 20 traveling-wave tubes will be fabricated for life tests. The test program for these 20 traveling-wave tubes follows.

The tubes will be subjected to a 2000-hour power aging test at laboratory ambient conditions. Tube parameters will be monitored at selected intervals during this 2000-hour test. The tubes will then be subjected to the expected flight environments consisting of vibration, shock, and thermal-vacuum. After the successful completion of the above environmental tests,

ten of the tubes will be subjected to a life test that will be conducted at laboratory ambient conditions. Five will be operated at the minimum expected temperature, and the remaining five will be operated at the maximum expected temperature.

The results of this test program may reduce the number of tubes that will be life tested as specified in the traveling -wave tube test plan for flight-quality tubes submitted in the April 1963 Advanced Syncom Monthly Progress Report.

#### Pyrotechnic Devices

Tests on pyrotechnic devices will be conducted on a lot acceptance basis. One homogeneous lot of devices is purchased and used both for the test program and flight spacecraft. Bruceton-type tests will be conducted on the test samples. Details of environments for each sample have not yet been determined. However, this will be accomplished well in advance of any actual tests which cannot occur until the lot purchase is received.

#### Test Program

Two valves shall be subjected to one cycle of each of the following qualification level environments:

- 1) Shock and vibration.
- 2) Thermal-vacuum and spin.
- 3) Boost acceleration.
- 4) High-temperature operation.
- 5) Low-temperature operation.

Each valve shall be actuated a minimum of 100 times during each environmental test. It is not necessary that the units be operated with propellants.

Before and after each environmental cycle the units will be tested for pulse characteristics and for leakage. The actual solenoid drivers will be used to excite the valves for all operational tests. Transient effects and power input will be measured as a function of temperature.

#### Bipropellant Injector Solenoid Valves Assemblies, Phase II

The injector solenoid valves (eight per spacecraft), control fuel, and oxidizer flow to the thrust chambers.

### Applicable Documents

- 1) Bipropellant Reaction Control Procurement Specification X254044, Revision A.
- 2) Injector Valve Qualification and Acceptance Test Plan

### General Requirements

- 1) The test valves selected shall be characteristic and identical in configuration to all flight production units and shall be assembled, inspected, tested, and handled in the same manner as all flight production units.
- 2) The performance tests conducted prior to, during, and after environmental testing shall be in accordance with those specified during qualification and acceptance.

### Test Program

Test Samples. A representative sample of 16 valve assemblies will be selected for the test program. Acceptance tests will be initially performed on all test specimens in accordance with the acceptance test specifications. Subsequently, these will be divided into four groups of three assemblies each for qualification tests, the remainder to be used in the extended life test.

Acceptance Tests. The 16 valves shall be divided into four groups of four each and subjected to one cycle of each of the following acceptance level environments.

- 1) Shock and vibration.
- 2) Thermal-vacuum and spin.
- 3) Boost acceleration.
- 4) High-temperature operation.
- 5) Low-temperature operation.

Parameters of the test cycle environments shall be as specified by the Syncom II unit acceptance test plan and appropriate performance tests conducted prior to, during, and after exposure.

The four groups of valve assemblies shall be tested in the following order relative to environments listed above:

| <u>Group</u> | <u>Environment</u> |
|--------------|--------------------|
| 1            | 1, 2, 3, 4, 5      |
| 2            | 1, 2, 3, 5, 4      |
| 3            | 5, 1, 3, 4, 2      |
| 4            | 2, 5, 4, 3, 1      |

The actual flight propellants will be used when the tests call for operation under load.

Upon completion of acceptance testing, one valve assembly from each group will be placed on an extended life test; the rest will undergo qualification test cycling.

Qualification Test Cycling. Twelve valve assemblies, upon completion of acceptance tests, shall be subjected to the environments listed in the acceptance tests according to the plan devised, except that qualification levels will be used.

Subsequent to group qualification testing, one valve assembly from each group will be placed on an extended life test. The remaining valve assemblies shall be subjected to additional cycles of each environment as follows:

| <u>Group</u> | <u>Environment</u> |
|--------------|--------------------|
| 1            | 1                  |
| 2            | 2                  |
| 3            | 3                  |
| 4            | 4 and 5            |

Extended Life Test Program. Four valve assemblies (two from acceptance test and two from qualification test) shall be subjected to pulsing at expected low temperatures. Four valve assemblies (two from acceptance test and two from qualification test) shall be subjected to pulsing at expected high temperatures.

The valves shall be pulsed for 25,000 operations over a 30-day period. The pulsing operation will include at least two periods of 100 milliseconds ON, 500 milliseconds OFF, and two periods of 14 minutes ON.

Following the environmental life tests, valve assemblies will be operated under ambient conditions and pulsed for 10-second periods once a day for at least 1 year.

The actual flight fluids will be used during life tests.

Complete functional tests shall be conducted at specified intervals during both phases of the extended life tests.

## MATERIALS TECHNOLOGY

Three types of solar panel substrates were made and evaluated in vibration. Two were configurations proposed by Hughes, one by NASA. One of the Hughes designs and the NASA design developed large vibration deflections with structural failures resulting. The other Hughes design, somewhat stiffer in construction, had a shear failure in adhesive bonding at an edge, but not of a major magnitude; this configuration will be used. The methods of mounting all of the panels during this testing subjected them to types of deflection which will not be experienced when mounted on the spacecraft.

The bipropellant control system proposal was analyzed; and design and drawings will be evaluated further.

The potting material used in the Syncom I traveling-wave tube may not be adequate for the higher operating temperature of the Syncom II tube. A new formulation of a silicone base with alumina filler is excellent in all respects except outgassing in vacuum. It cannot be adequately "vacuum cleaned," but it is believed that the ends of the tube can be hermetically sealed by discs of epoxy-glass. A tube of this construction is being prepared for testing.

A flexible potting compound for connectors was developed and a process specification for its use is being prepared. It does not outgas in vacuum. This compound can also be used on new Syncom I harnesses to replace the more rigid type initially used for this purpose.

Data behavior of three types of thermal control coatings were analyzed; a plan was prepared for continuing and extending these tests to predict quantitatively the 3- to 5-year behavior of such materials. A more comprehensive plan for a similar study of other materials was completed and submitted for consideration.

Some magnetic parts for power supplies were designed and samples for breadboard trials were constructed. This work continues as circuit requirements become available.

Specifications were prepared for additional frequency-controlled crystals required.

Evaluation of new types of RF connectors was started.

Evaluation of several new types of wire and soldering techniques was started.

## 7. SPACECRAFT SUPPORT EQUIPMENT, RELATED SYSTEMS TESTS, AND INTERFACES

### INTERFACE SPECIFICATIONS

Interface documents will be prepared to define interfaces among the major elements of the Syncom II system as follows:

- 1) Launch System/Hughes: The launch system defines the spacecraft and spacecraft-GSE interfaces with the launch vehicle system and the launch complex.
- 2) Apogee Motor System/Hughes: The spacecraft and spacecraft-GSE interfaces with the apogee motor and the apogee motor ground support equipment (GSE) are defined.
- 3) Bipropellant System/Hughes: The spacecraft and spacecraft-GSE interfaces with the bipropellant system and the bipropellant GSE are defined.
- 4) Ground Communication Terminal/Hughes: The communication terminal interface with the spacecraft, the telemetry and command stations, and the GSFC is defined.
- 5) Telemetry and Command Station/Hughes: The telemetry and command station interfaces with the spacecraft, the communication terminal, and GSFC are defined.

The proposed outline of the Launch System/Hughes Interface document is given below as an example of the general format and content of the interface reports.

- 1) Introduction  
(Statement of responsibilities Hughes Aircraft Company, NASA, LMSC)
- 2) Purpose  
(Identify interface problems and provide a current list of applicable references)

- 3) Launch System Performance Requirements
  - a) Prelaunch checkout and servicing requirements
  - b) Launch and parking orbit requirements(Describe launch system requirements and provide a current list of applicable references)
- 4) Launch System Description
  - a) Launch vehicle system
  - b) Prelaunch checkout and servicing system(Describe launch system design and provide a current list of applicable references)
- 5) Launch Vehicle/Spacecraft Electrical Interfaces
  - a) Spacecraft/umbilical
  - b) Spacecraft GSE/launch vehicle GSE
  - c) Spacecraft GSE/NASA(Describe electrical interfaces and provide a current list of applicable references)
- 6) Launch Vehicle/Spacecraft Mechanical Interfaces
  - a) Launch vehicle/spacecraft
  - b) Spacecraft/umbilical
  - c) Spacecraft GSE/launch GSE(Describe mechanical interfaces and provide a current list of applicable references)
- 7) Launch Base Facility Interfaces
  - a) Space requirements
  - b) Power requirements
  - c) Air-conditioning requirements
  - d) Communication requirements(Describe Hughes Aircraft Company/launch base facility interfaces and provide a current list of applicable documents)
- 8) Launch Base Personnel Interfaces
  - a) Operator training
  - b) Station manning(Describe personnel responsibilities and provide a list of applicable references)



9) Documentation Interfaces

- a) Test plans
- b) Test procedures
- c) Specifications

(Describe document requirements and responsibility of document preparation and integration)

## GROUND SUPPORT EQUIPMENT

The following aspects of ground support equipment were pursued during this report period:

- 1) A program definition was prepared which delineates the tasks to be undertaken and their status at the completion of the present contract.
- 2) The test console preliminary layout and description was undertaken with the definition of controls, functions, and displays being outlined.
- 3) The ground support equipment system block diagram revision based on a subsystem philosophy was undertaken.
- 4) Studies of data handling requirements were initiated.

### Program Management and Definition

The general areas to be undertaken during the present contract are outlined:

| <u>Tasks</u>                      | <u>Status at Completion Period</u>                             |
|-----------------------------------|--|
| Standard equipment specifications | Specifications completed and issued                            |
| Block diagram updating            | Updated diagram, interface specifications completed and issued |
| Rack and console layout           | Complete layout and inter-connection specifications            |
| Breadboard fabrication            | Complete breadboard of selected components                     |

| <u>Tasks</u>          | <u>Status at Completion Period</u>   |
|-----------------------|--|
| Data handling studies | Date flow chart, analysis of data rates, data format specifications and commercial EQ selection complete |
| Test plans            | Complete test plans  |
| <u>Test Console</u>   |  |

The control console is conceived as containing all the required command and display functions to allow the test engineer to conduct and control all system tests from his position at the console. The major items required and the displays required have been identified and work is progressing on the major-item layout.

Table 7-1 contains a list of the major units to be found on the console. The significant Hughes-built units on the control console are the command signal generator, the telemetry display panel, and the ground synchronous controller. Work has begun on each of these items.

Table 7-2 is a preliminary list of the design objectives for the command signal generator; work is currently under way on this unit, both circuit design and product layout. It is anticipated that a logic design and block diagram for the command signal generator will be completed during the next report period. A preliminary layout of the front panel should also be completed during this time period.

Table 7-3 contains a preliminary list of the displays to be included on the telemetry display panel. It is intended that the telemetry display panel will be used to display both telemetered information and selected locally generated information.

The telemetry processor will be capable of being programmed so that any of 11 different pieces of information may be displayed on any of the various display units.

Currently, the telemetry display panel is envisioned as consisting of seven digital readouts and five analog readouts. The digital readouts will display information which has been converted to engineering units, i. e., tank pressure will be displayed in pounds per square inch. Work is now under way on the logic design of the telemetry processor as well as the telemetry display panel.

Effort is also progressing on the logic design of the ground synchronous controller. It is anticipated that a block diagram of this unit will be completed during the next report period.

TABLE 7-1. PRELIMINARY LIST OF CONSOLE EQUIPMENT

- 1) Command signal generator
- 2) Telemetry displays
- 3) Synchronous controller
- 4) Time displays
  - a) Greenwich mean time
  - b) Test time
- 5) Power controls
  - a) Spacecraft power
  - b) Spin machine power
- 6) Communications
  - a) Intercom jacks (2)
  - b) Telephone
- 7) Speaker
- 8) Miscellaneous indicator (20 on-off bulbs on one panel)
  - a) Spacecraft power, external and internal
  - b) Telemetry receiver lock, (2)
  - c) Recorders (4)
  - d) Communications test equipment loop lock, (4)
- 9) Modulator meter (in parallel with normal modulation indication)
- 10) On-off cycle and running time meters for status indicators of command signal generator (not on panel)

TABLE 7-2. DESIGN OBJECTIVES FOR COMMAND SIGNAL GENERATOR

I. Required Front Panel Controls and Displays

A. Controls

- 1) Commands will be selected by inserting the proper address and command into a series of thumb-wheel switches.
- 2) Manual insertion of the digital commands should be provided, perhaps as a set of toggle switches or thumb-wheel switches behind a flap.
- 3) A mode-select switch shall be provided so that either the primary (shift) mode or the secondary (count) mode may be selected. Each mode would accept a command input from either the pushbuttons (I. A. 1) or the switches (I. A. 2).
- 4) Three modes of control of the execute pulse circuitry are required.
  - a) A one-shot pulse of variable width (10 to 150 ms).
  - b) Execute and hold (no decode clear).
  - c) A constant-execute pulse.
  - d) An execute pulse controlled by the ground synchronous controller.
- 5) Antenna position control should be automatically selected by thumb-wheel switches (from 0 to 360 degrees).
- 6) Lock control which would keep the command generator in safe.
- 7) Self-test capability.

B. Displays

- 1) Indication of status of spacecraft electronic subsystem; implement memory from commands or from telemetry. If telemetry is used, some backup mode must be provided.
- 2) Automatic verification of commands; display in the same number inserted by the thumb-wheel switches used to generate the command.

TABLE 7-2 (continued)

- 3) Both sent and verified commands displayed digitally so that a comparison can be made
- 4) Spacecraft status indicators operated from both sent and verified commands.
- 5) Display of telemetered "enable" signal.

## II. Operational Characteristics

- A. Automatic execute-clear sequence in the execute pulse mode (see Paragraph I. A. 4. a).
- B. Selectable "decoder clear" command.
- C. A 3-minute interlock between "TWT Filament On" and "High Voltage On" command, light to indicate "Ready" and an override switch.
- D. Direct jet commands execute only through synchronous controller circuitry.
- E. An output to recorder of spacecraft status indication.
- F. On-off cycle counter and running time meter associated with selected status lights. These displays are not required on the panel and may be under the shelf (out of normal view).

Equipment Rack Layout. Effort is under way on equipment rack layout. It is anticipated that during the next report period an initial layout of all the racks will be completed as well as basic interconnecting cabling.

Communication Transponder Test Equipment. Figure 7-1 is a block diagram of the communications transponder test set. A breadboard model of this unit will be produced during the period of the contract.

## System Block Diagram

Further study produced minor modifications to the Syncom II test station block diagram contained in the supplement to the Summary Report. These investigations have yielded voltage and impedance levels in the telemetry subsystem further definitizing that portion of the diagram. The basic test philosophy is unchanged provided that in-laboratory satellite checkout equipment is suitable for launch site operations (Atlantic Missile Range) with minor substitutions. (Substitutions involve spin machine controls, spacecraft external power, etc.) Additional information that can be inserted on the block diagram follows:



TABLE 7-3. DESIGN OBJECTIVES FOR TELEMETRY  
DISPLAY PANEL

- 1) Five-digit projection displays (four numeric, one for title and units) will be used to display the following:
  - a) Unregulated bus voltage
  - b) Battery voltage
  - c) Current (solar panel or ground supply)
  - d) Fuel tank pressures
  - e) Travelling-wave tube and solar panel temperature
  - f) Antenna position (spacecraft)
  - g) Antenna position (updated on the ground)

Each display will have a selector switch to select which bus, battery, etc., will be monitored.

- 2) Meters to display analog signals such as:
  - a) Traveling-wave tube power
  - b) Received signal
  - c) Telemetry power
- 3) Indicator to show which telemeters are on
- 4) Switches to select which telemeter is used for telemetry data display
- 5) Status lights for:
  - a) Spacecraft identification
  - b) Warning for battery/solar cell operation
  - c) Quadrant identification
  - d) Telemetry processor lock

- 1) Synchronous controller output to the command generator, 0 to 7 volts
- 2) Input to the synchronous controller from the processor, 0 to 7 volts
- 3) Output of the command code generator to the modulator is 0.5 volts for 100 percent modulation into an input impedance of approximately 500 ohms at a frequency of 350 to 718 kc
- 4) Command generator inputs from the processor are 0 to 7 volts (200 bits per second NASA system)
- 5) The processor inputs from the telemetry receivers are 0 to 5 volts. The discriminator output, 0 to 5 volts; the decoder output, 0 to 7 volts, and the low pass filter and detector output, 0 to 5 volts

The semiautomated patch panel concept was further investigated with visits from various vendors who displayed state-of-the-art equipment. The diagram is at present being modified to indicate the various subsystems and an interface specification.



## 8. SPACECRAFT HANDLING EQUIPMENT

Design was initiated during the latter portion of this period on the mobile assembly fixture. Design of the clamp attachment to secure the Syncom to the fixture is continuing.

Location of the rotational axis of the fixture is pending, contingent upon final establishment of the weight and center of gravity of the spacecraft.

A preliminary design study of the requirements for the mobile test stand was initiated.

## 9. NEW TECHNOLOGY

### HORIZON SCANNERS

The supplement to Summary Report, April 1963 Monthly Progress Report, Pages 5-34 through 5-45, describes the method for determining the angle between the spin axis and the equatorial plane by the use of earth sensors. This method is believed to be new. Although the use of earth sensors for this purpose is not new, the particular technique developed is considered unique. Automatic inhibit-logic circuitry to preclude erroneous interference signals from the sun and moon is now under development and will be reported when completed.

The technique is considered an innovation rather than a patentable item.

### GENERAL

Other new technology reports have been prepared and are presently being reviewed for patent feasibility. These technologies will be included in a future report.

## 10. PROJECT REFERENCE REPORTS

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## APPENDIX A. COMMAND DECODER DESIGN REVIEW

The proposed Syncom II command decoder is a three-tone, dual-mode system which, during operation, requires that only one tone be present at a time. The system requires only one input from the command receiver; and it furnishes 128 individual outputs, each of which corresponds to a spacecraft command signal. A complete block diagram of the decoder is shown in Figure A-1.

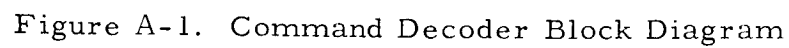
Under normal operating conditions the decoder responds in the primary mode to an input which is frequency-shift-keyed (FSK) modulated. The essence of this type of modulation is that the message is sent in with two tones, one to signify a logical one, the other a logical zero. The message is shifted into the command register with a bit sync clock signal which amplitude modulates both signal frequencies. Thus, while only one tone is present at any one time, the clock is always present.

The secondary mode of the decoder responds to a 100 percent pulse amplitude modulated input which corresponds to interrupting the message tone at regular intervals. This mode requires that only one tone channel be used to send the entire message. The command register counts the interrupts and so accumulates the desired message.

### DECODER OPERATION

Prior to sending a message, both message tones must be stopped for a short period of time ( $< 10$  milliseconds) during which timing circuitry is allowed to recover. It is assumed that one tone or the other has been on when a message is not being sent. This assumption may not necessarily hold, but it will if the command system is to be made secure from outside influences when it is not in use. If it is desired to operate in the primary mode (called shift mode), the zero's tone is then turned on and left on until the message is to be sent.

Shifting to the one's tone will cause word sync to be detected, and the next seven bits will address enable power. Enable power corresponds to a latching switch which supplies power to the execute circuitry when enable is turned on. When it is off, the presence of an execute tone means nothing to



the decoder. If the correct address is read after the seventh bit, enable power will come on. The eighth through fourteenth bits correspond to the command which will ultimately be executed, provided it is verified correctly by telemetry. After the fourteenth bit has been detected, the input to the command register is disabled and any further inputs to the decoder will be ignored by the command register. When verification is completed both message tones will be terminated and the execute tone sent, causing the command to be executed.

The operation of the secondary mode (count mode) is started in a manner similar to the primary mode. Both tones are interrupted for a short period of time, and then either tone is turned on and left on for some minimum length of time. The tone which has been present is then interrupted again and, at this time, the decoder will shift into the secondary mode. After the tone has been absent for a period of time, either tone is sent with the required modulation. (The tone not being used will be off.) The command register will accumulate the enable address, and when the desired number of pulses has been sent the tone is interrupted again for a given length of time. During this time, the enable will read the contents of the command register, and, if the address is correct, the enable will come on. At the same time the command register will be reset to zero. The modulated input tone is then reestablished and the command will be counted into the command register. After the desired command is accumulated the tone is turned off, and a short time later the input to the command register is disabled. Following verification by telemetry the execute tone is established, and the command is executed. Following execution of a command, the decoder can be completely cleared by applying a continuous uninterrupted message tone. This applies to either mode.

If it is found for either mode of operation that the contents of the command register do not correspond to the message sent from the ground, it is possible to clear the decoder completely. This is done by sending a continuous, uninterrupted message tone when in the count mode, or by doing the same thing after interrupting both tones for a short period of time when in the shift mode.

#### Explanation of Functional Circuit Blocks

The nomenclature used below to identify the circuits described is defined as follows:

- 1) The C which precedes the number indicates that the circuit is part of the command decoder.
- 2) The first numeral identifies the type of circuit where:

1xx corresponds to a flip-flop  
3xx corresponds to an amplifier or shaper  
5xx corresponds to an inverter



6xx corresponds to a circuit that is not a standard type  
7xx corresponds to a filter  
8xx corresponds to a special type of gating

- 3) The second and third numerals designate a particular circuit.

#### C700, C701, C702

The tone filters C700, C701, and C702 are narrow-bandpass filters which will be designed to respond to three individual frequencies. C702 corresponds to the execute tone filter, C700 corresponds to what shall be called the zero's tone filter, and C701 corresponds to the one's tone filter. The input to the filters comes from the command receiver.

#### C300, C301, C302

The bias detectors and shapers C300, C301, and C302 have as their inputs the tone filters C700, C701, and C702 respectively. The bias detectors and shapers function as their name implies. They shape the outputs of the filters into waveforms which are suitable for digital use. They have a fixed bias level which, for a continuous sine wave amplitude modulated input, will put out a fixed unvarying voltage at some level greater than ground (+24 volts). When a tone is completely absent the output shall be close to ground. When the input is 100 percent amplitude modulated with a pulsed type of waveform at a duty cycle which will be less than (preferred) or equal to 50 percent, the output will be pulsed also with an envelope which corresponds to the envelope of the input. (The latter case applies to the secondary mode.) The output of C302 will respond in a manner similar to C300 and C301 depending on the waveform of the execute.

#### C606

The audio detector and shaper, C606, has as inputs both C700 and C701. The output is a square pulsed waveform regardless of the mode of operation. Waveform outputs of C300, C301, and C606 are shown in Figures A-2 and A-3 as a function of their inputs C700 and C701.

#### C603

The read tone present circuit, C603, has as its inputs  $\overline{C300}$  and  $\overline{C301}$ . The output of C603 depends on the logical "AND" combination of the inputs. If both tone inputs to the decoder are absent long enough for the circuit to recover, the output will reach a logical one (high) state and stay there until a single tone (or the same logical function using both inputs, i. e., one always high while the other is low) has been present for a fixed length of time, and at this time the output will go to a logical zero (low) state with a fast fall time.

### C604, C605

The reset circuits C604 and C605, called respectively the "command register reset" and "all register clear" circuits, are triggered by negative voltage transitions at the input. The output is a short pulse which goes slightly negative during the pulse interval and resets the desired individual flip-flops or registers of flip-flops. C604 is triggered by C603 (as is C605) and also by negative transitions of  $\overline{C108}$  (when C109 is low) and enable power, C601, which are discussed below.

### C602

The read tone interrupt circuit, C602, is controlled by the logical "OR" combination of C300 and C301. When either tone has been present for some period of time such that C602 has time to recover to the logical one state, then the absence of both tones for a reasonably long period of time (after the recovery period) will cause C602 to fall to logical zero with a fast fall time.

### C108, C109

The interrupt counter, C108 and C109, consists of flip-flops. C108 is a clocked pedestal gated flip-flop and C109 is a standard pedestal gated flip-flop the input of which is the flip-flop output of C108. The "T" input to C108 is controlled by  $\overline{C110}$  which is discussed below. The clock for C108 is derived from C602. As the name implies, the interrupt counter counts the negative transitions of C602, which correspond to interrupts in the input tone provided  $\overline{C110}$  is in the logical one state. The interrupt counter is reset to zero by C605 which is controlled by C603 (discussed above).

### C110

The word sync flip-flop, C110, is used to detect word sync when no interrupts have occurred prior to detection. When C108 and C109 are low and the proper configuration comes up in the command register the inverter C501 will set C110 in the logical one state indicating that word sync has been detected. C110 is reset to zero by C605.

### C609

The bit counter reset, C609, resets the bit counter with a short duration pulse which goes to a negative voltage during the pulse interval. The pulse is initiated by a negative transition of  $\overline{C110}$ .

### C111 through C114

The bit counter, C111 through C114, is made up of four pedestal gates flip-flops. It is used to count the bits which are being shifted into the command register after word sync has been detected. (Since the bit counter is reset to zero by  $\overline{C110}$ , it is ready to count from zero after word sync has been detected.)

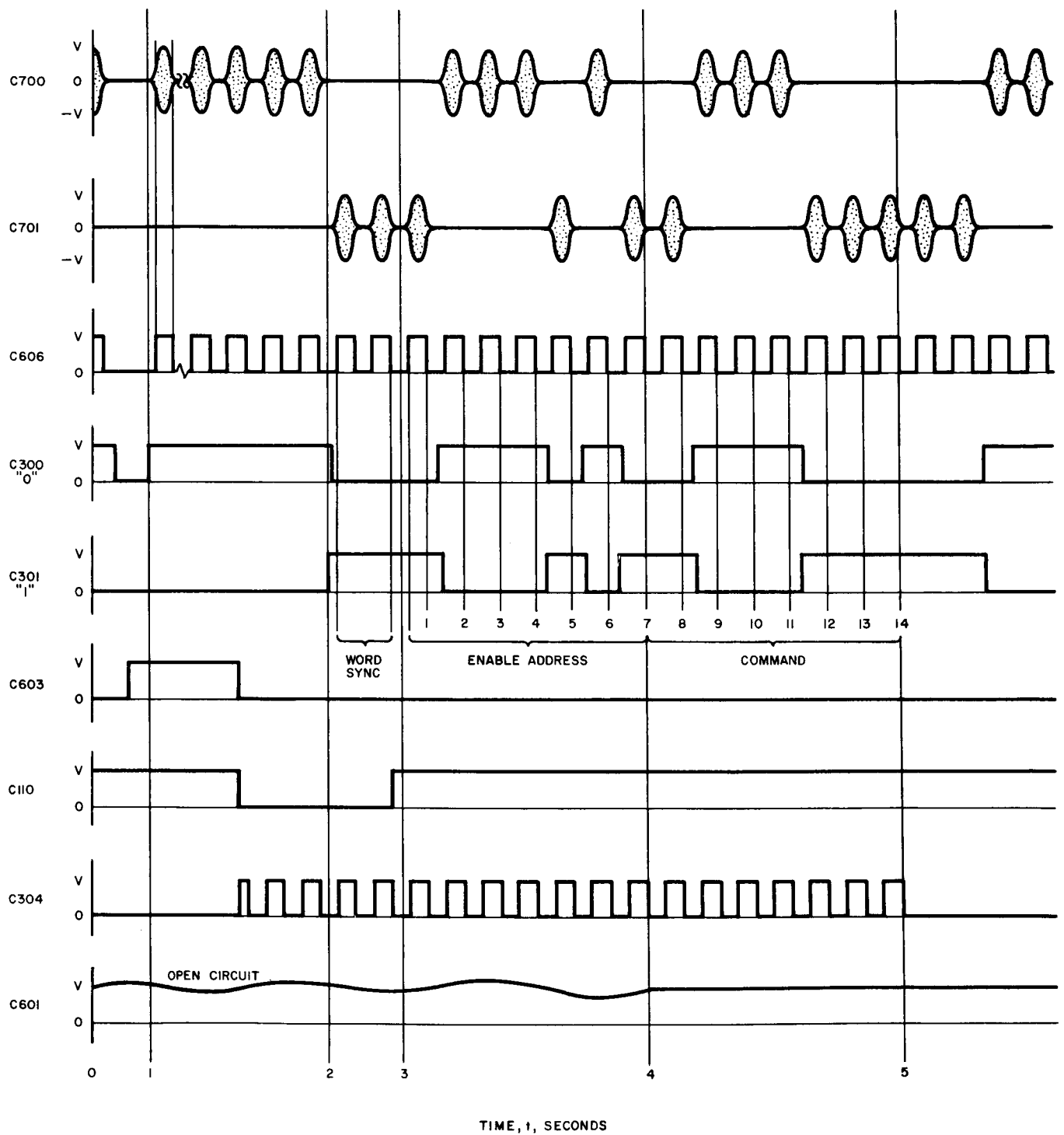


Figure A-2. Command Decoder Waveform Shift Mode

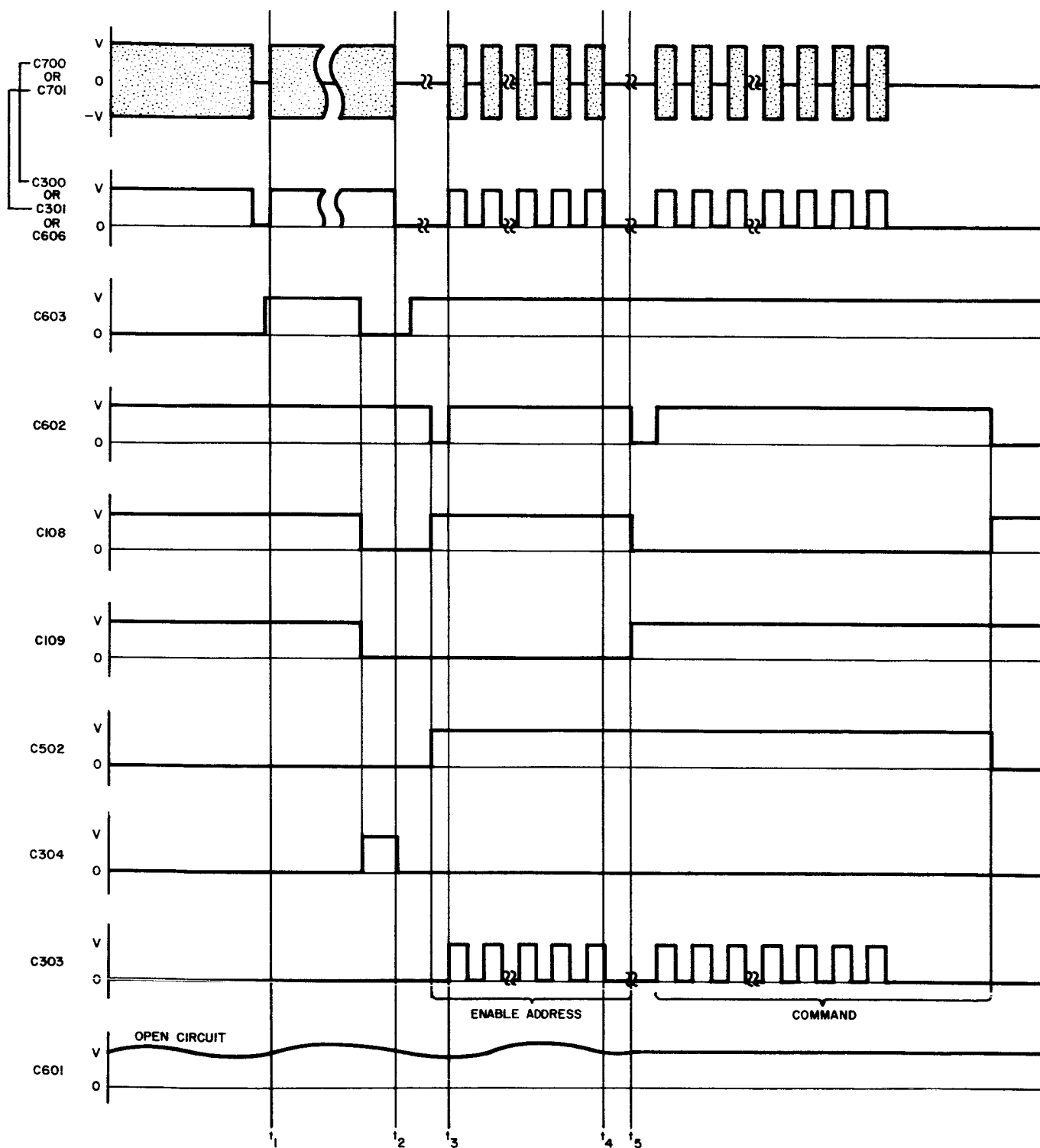


Figure A-3. Command Decoder Waveform Count Mode

### C303, C304

The count mode clock, C303, and the shift mode clock, C304, are simple "AND - OR" amplifiers which are controlled by input gating. When the decoder is in the primary mode C303 is in the low state and C304 provides the shift clock. When the decoder is in the secondary mode, C304 is in the low state, and C303 provides the count clock. A more detailed discussion of the control gating is given below in the detailed mode description.

### C601

The enable power circuit, C601, is a latching switch which provides power to bias detector and shaper C302, and the matrix power (C600) circuits. The enable power circuit is turned on by a correct address present in the command register at the proper time, and it is reset by C603 (discussed above).

### C101 through C107

The command register, C101 through C107, consists of seven flip-flops which can, through the use of digital switches and special gating, be used to accumulate a desired message by counting or shifting in the input.

### C608

The count inhibit power circuit, C608, is used to provide power to gating which prevents incongruities from occurring in the command register, when operating in the shift mode, due to the presence of the count mode circuitry.

### C800

The diode matrix, C800, is an arrangement of diodes which provides all possible combinations (128) of the binary outputs of the command register. The power to the gating resistors of the matrix is controlled by C600, designated the matrix power circuit.

The operation of C302 was discussed above. The output is high when an execute tone is present provided that the enable has been turned on.

### C600

The matrix power circuit, C600, works in conjunction with C302 in that it provides power to the diode matrix when an execute tone is present provided that enable power is on. When an execute tone is present all of the gates in the matrix will have power applied to them and the required command signal will be distributed throughout the spacecraft.

## Detailed Decoder Operation

The explanation of decoder operation can more easily be accomplished by reference to the timing waveforms in Figures A-2 and A-3, which apply to the shift and count modes respectively.

It is assumed, regardless of the mode of operation to be chosen, that initially one message tone (as opposed to the execute tone) will be on, although this may not necessarily be true.

### Primary (Shift) Mode Operation

Output waveforms discussed below are referenced to Figure A-2. Prior to sending a message to the spacecraft, the tone which has been on must be interrupted for several milliseconds. Following this period of time the zero's tone is turned on and the output of C700 will appear as shown between  $t_1$  and  $t_2$ . A minimum of 15 bit sync cycles must elapse before the word sync is sent. The total interrupt of the message tone allows C603 to recover, and between  $t_1$  and  $t_2$  the fall in the output of C603 resets all of the decoder registers to zero with the exception of the bit counter, C111 through C114. Operation in this mode, after  $t_2$ , never allows C603 to recover again since, henceforth, a tone will always be present at one input or the other.

At the time  $t_2$ , the states of the mode control circuits in the decoder will be as follows: The logical configuration of the interrupt counter, C108 and C109, will cause C502 to be low. As a result, C608 will be providing power to the count inhibit circuitry and C303 (count mode clock) will be disabled. The shift mode clock C304 will be operating and zero's will be shifted into the command register until word sync is detected.

Between  $t_2$  and  $t_3$  several one's are sent and when the right number is detected C501 will go low (at  $t_3$ ) setting the word sync flip-flop, C110, high. When C110 goes high, the bit counter will be reset to zero by C609 and the input to the interrupt counter will be disabled. After  $t_3$  the bit counter will count the number of clock pulses put out by the shift mode clock, C304.

Starting at  $t_3$  the command register begins to accumulate the enable address and at  $t_4$ , seven bits after the detection of word sync, provided the address is correct, enable power will be turned on by the bit counter in combination with the contents of the command register.

At  $t_4$  the command register begins to accumulate the command and as soon as the seven command bits have been accumulated, the shift clock, C304, will be turned off and its output will stay at ground. This occurs at  $t_5$ ; i.e., 14 bit times after the detection of word sync, the shift clock is disabled.

As soon as verification (via the telemetry link) is complete, the message tone, if it is on, is turned off and the execute tone is turned on. C302 will then turn on the matrix power circuitry and the command will be executed within the spacecraft (provided, of course, that enable power is on).

After the execution technique is completed, either tone can be turned on and the command register will be cleared so that undesired executions cannot occur accidentally. Also, the decoder will be secure against any possible outside influences.

### Secondary (Count) Mode Operation

Output waveforms discussed below are referenced to Figure A-3. As in primary mode operation, the continuous tone which has been present for some time is interrupted and C603 is allowed to recover. At time  $t_1$  either tone can be turned on; and, sometime later, all of the registers in the decoder again, with the exception of the bit counter, will be reset to zero. The interrupt time is of the same order of magnitude as that for the primary mode. At time  $t_2$  the input tone is interrupted again, and a short time later (at  $t_3$ ) C602 goes low setting C108 high.

When C108 goes high, the input to the word sync flip-flop, C110, is disabled and C110 will remain in the reset condition. Also, when C108 goes high, the output of C502 will go high, disabling the count inhibit power circuitry and enabling the input to the count mode clock, C303. The input to the shift mode clock will be disabled at the same time, and its output will stay at ground. The command register will also be reset to zero by C604.

At  $t_3$  the decoder is ready to count short interrupts in the input tone. When the tone is reinstated it will be pulse amplitude modulated and the command register will count the number of pulses until time  $t_4$  when the enable address has presumably been counted into the command register. At this time the input is interrupted again for a reasonably long period of time and C602 will go low, causing C108 to go low and C109 to go high. This will occur at  $t_5$ , and at this time the combination of C109, C108, and the command register output configuration will cause enable power to come on provided the correct address is present in the command register. When the enable power comes on, the command register will be again reset to zero by C604.

After the command register has been reset at  $t_5$  the pulse amplitude modulated input tone is resumed and the command is counted into the command register. When the right number of counts has been accumulated the tone is again interrupted, and a short time later C602 will make another negative transition, causing C108 to go high and so disabling the count mode clock and causing it to remain near ground. Following verification, the command is executed as in the primary mode.

### Single-Mode Command Decoders

Reference may be made to the circuit block descriptions and to the timing waveforms where applicable.

### Count-Type System

Figure A-4 is a block diagram of a single-mode, count-type, command decoder. It requires two tone channels, an execute tone and a message tone. This decoder responds to a 100 percent pulse modulated input message tone. Its operation is similar to that of the count mode in the dual-mode decoder.

Prior to the time when a message is to be sent, the message tone may or may not be present. If it is present, the tone must be interrupted for a short period of time during which the read tone present circuit will recover; and its output will go high. If the message tone has not been present this circuit will already have recovered. After the read tone present circuit has been allowed to recover, a continuous message tone is sent for a reasonably long period of time; and during this time, the command register and the interrupt counter will be reset to zero.

After these registers have been cleared the tone is interrupted; and a short time later the output of the read tone interrupt circuit will go low and set flip-flop C108 of the interrupt counter high. At the same time the command register will be cleared again.

At this time the message is sent. The command register, which is a pedestal gated counter, begins to count the negative edges of C504, which correspond to the times when the input tone falls to zero. When the desired number of pulses have been sent from the ground, the message tone is interrupted again; and after some time, the output of the read tone interrupt circuit goes low again, causing flip-flops C108 to go low and C109 to go high.

When this occurs the input to the enable power circuit is such that if the contents of the command register correspond to the required configuration, the enable power will come on. Power will then be applied to the execute circuitry; and at the same time, the command register will be cleared.

The message tone, with the required modulation, is then reinstated; and the command register will begin to accumulate the command. When the desired number of pulses have been sent from the ground, the tone is interrupted again; and after some period of time, the output of the read interrupt circuitry will go low again causing C108 to go high. At this time, the input to the command register will be blocked; and any further inputs from the message tone bias detector, C301, will be ignored.

The contents of the command register are then verified via telemetry; and if its configuration corresponds to the desired command, the execute tone is turned on. The presence of the execute tone will, provided enable power is on, cause power to be applied to the resistors of the diode matrix, and the command will be executed within the spacecraft.



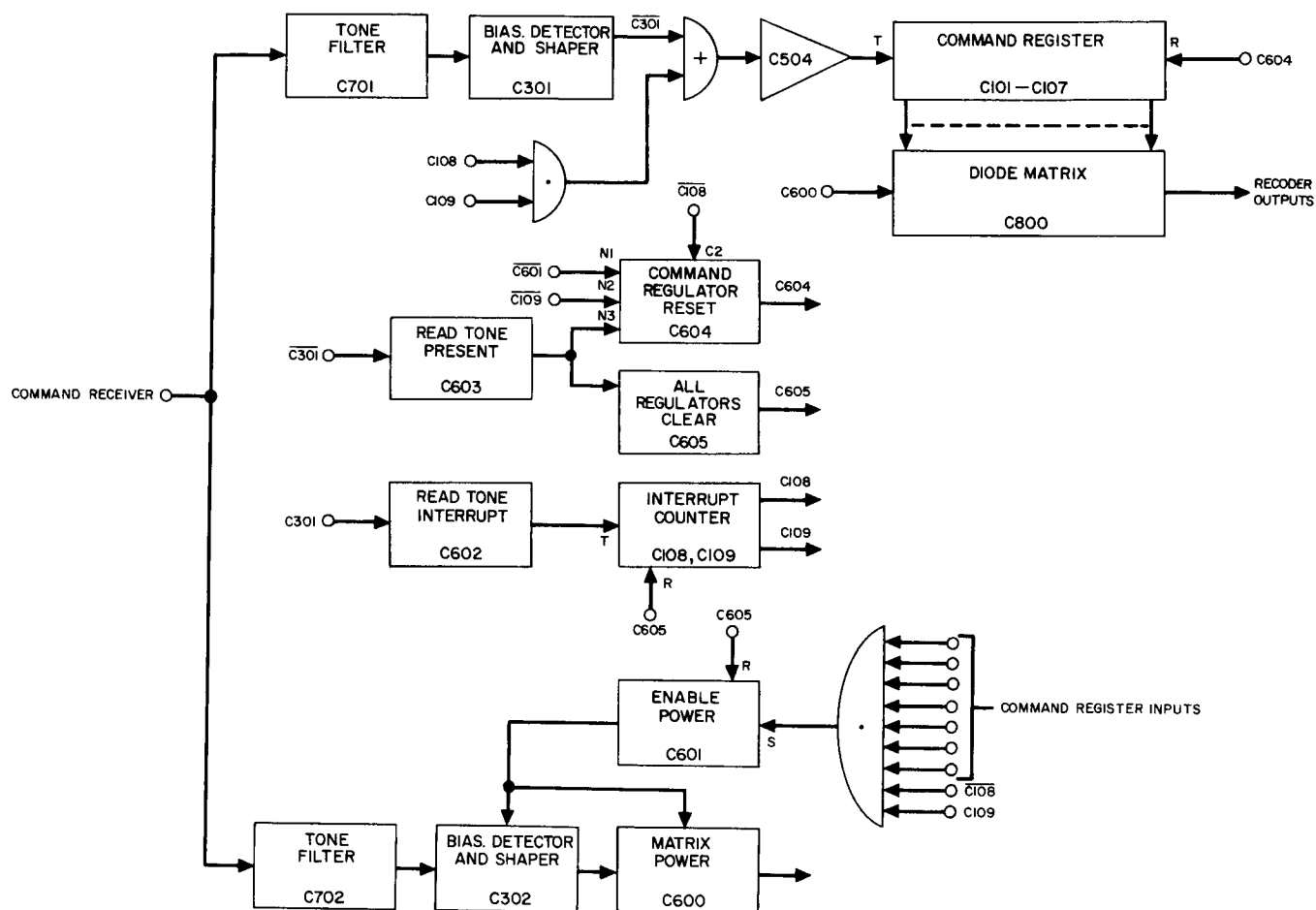


Figure A-4. Command Decoder Count Operation Block Diagram

Following execution, the execute tone is turned off and a continuous message tone is sent. The read tone present circuitry will then cause the command register to be reset to zero, eliminating the possibility of further undesired executions.

### FSK-Type System

Figure A-5 is a block diagram of a single-mode FSK type of command decoder. It requires two message tone channels and an execute tone channel. This decoder responds to an FSK modulated input. Its operation is similar to that of the shift mode in the dual-mode decoder.

Prior to the time when a message is to be sent, either (but not both) of the message tones may have been on or both may have been off. It is required, as in the case of the count mode discussed above, that both message tones be off for a short period of time prior to sending a message. After this requirement has been satisfied, the zero's tone is turned on and zeros will be shifted continuously into the command register. The presence of the zero's tone for some minimum period of time is required so that the output of the read tone present circuit may make a negative transition causing the word sync flip-flop and the command register to be reset to zero.

After this requirement has been fulfilled the entire message is sent without interruption. The first several bits of the message correspond to word sync. When the contents of the command register correspond to the required configuration, word sync will be detected, causing C501 to go low setting the word sync flip-flop high. When this occurs, the bit counter will be reset to zero by the bit counter reset; and it is ready to count the bits which are being shifted into the command register.

Following the detection of word sync, the next seven bits correspond to the enable power address. After the seventh bit, enable power will be turned on if the contents of the command register correspond to the correct enable address.

The next seven bits (eighth through fourteenth after detection of word sync) shifted into the command register correspond to the command.

After the fourteenth bit the input to the command register will be disabled automatically by the bit counter, and any further changes in the input message tones will not be detected.

At this time the contents of the command register are verified via telemetry; and if the configuration of the command register corresponds to the proper command, the execute tone is sent (after the message tones have been turned off). The presence of the execute tone will cause power to be applied to the diode matrix (provided that enable power is on), and the command will be executed within the spacecraft.



Following execution of the command, the execute tone is turned off; and either message tone is sent continuously for some period of time. The read tone present circuit will then cause the command register to be cleared, eliminating the possibility of any further undesired executions.

## FAILURE RATES

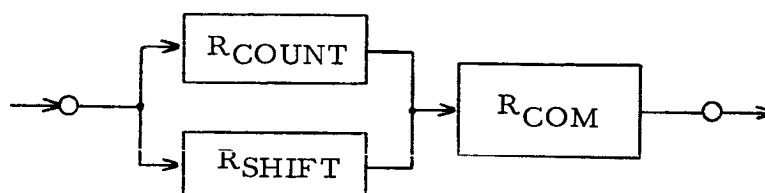
The failure rates used in the determination of the reliability of the command decoder were obtained from the R-67-2 handbook and from reliability personnel of Hughes Aircraft Company. Those which do not correspond to failure rates in the R-67-2 handbook are justified on the grounds of greatly reduced stresses from those specified in the handbook, and also on the premise that the parts which are concerned will be subjected to tests which should decrease the probability of their failure to a degree consistent with the rates used.

Carbon composition resistors — 0.001 percent/1000 hours.  
Carbon film resistors — 0.016 percent/1000 hours.  
Glass and mica and paper capacitors — 0.001 percent/1000 hours.  
Tantalum capacitors — 0.004 percent/1000 hours.  
Inductors — 0.002 percent/1000 hours.  
Diodes — Switching — 0.002 percent/1000 hours.  
          Zener — 0.015 percent/1000 hours.  
Transistors — 0.005 percent/1000 hours.

## RELIABILITY

### Dual-Mode System

The reliability of the dual-mode system is calculated on the premise that no redundancy is used. Even so it can be divided into three functional sub-subsystem blocks, as shown below for purposes of computing reliability.



The R<sub>COM</sub> block contains all circuitry which is common to and necessary for operation of either mode. The R<sub>COUNT</sub> block contains circuitry which is necessary for the operation of the count mode only and which will not affect operation of the shift mode in the event that it fails. The R<sub>SHIFT</sub> block contains circuitry necessary only for the operation of the shift mode, and any failure in this block will not constitute a failure of the count mode.

Parts breakdown by sub-subsystem block and failure rate:

|                                   | <u>Number of<br/>Components</u> | <u>Failure Rate,<br/>(percent/<br/>1000 hours)</u> |
|-----------------------------------|---------------------------------|--|
| $R_{COM}$ :                       |                                 |  |
| Carbon composition resistors      | 288                             | 0.288  |
| Carbon film resistors             | 20                              | 0.320  |
| Glass, mica, and paper capacitors | 39                              | 0.039  |
| Tantalum capacitors               | 11                              | 0.044  |
| Inductors                         | 8                               | 0.016  |
| Diodes                            |                                 |  |
| Switching                         | 410                             | 0.820  |
| Zener                             | 2                               | 0.030  |
| Transistors                       | 53                              | <u>0.265</u>                                       |
| Total $R_{COM}$ failure rate      |                                 | 1.822  |

$$MTBF R_{COM} = 55,000 \text{ hours} = 6.28 \text{ years}$$

|                                |    |              |
|--------------------------------|----|--------------|
| $R_{COUNT}$ :                  |    |              |
| Carbon composition resistors   | 20 | 0.020        |
| Glass and mica capacitors      | 28 | 0.028        |
| Diodes, switching              | 38 | 0.076        |
| Transistors                    | 2  | <u>0.010</u> |
| Total $R_{COUNT}$ failure rate |    | 0.134        |

$$MTBF R_{COUNT} = 746,000 \text{ hours} = 85.3 \text{ years}$$

|                                   |    |              |
|-----------------------------------|----|--------------|
| $R_{SHIFT}$ :                     |    |              |
| Carbon composition resistors      | 63 | 0.063        |
| Carbon film resistors             | 12 | 0.192        |
| Glass, mica, and paper capacitors | 52 | 0.052        |
| Tantalum capacitors               | 6  | 0.024        |
| Inductors                         | 4  | 0.016        |
| Diodes, switching                 | 55 | 0.110        |
| Transistors                       | 19 | <u>0.095</u> |
| Total $R_{SHIFT}$ failure rate    |    | 0.552        |

$$MTBF = 181,000 \text{ hours} = 20.6 \text{ years}$$

$$\text{Total system parts} = 1130$$

### Subsystem Reliability:

$$P(s) = e^{-t/MTBF} = 1 - q$$

where  $p(s)$  is the probability of success for some time  $t$ , and  $q$  is the probability of failure for that time.

$$P(s)_{\text{subsystem}} = P(s)_{\text{COM}} \left[ 1 - q_{\text{SHIFT}} q_{\text{COUNT}} \right]$$

For 1 year, considering one quadrant:

$$P(s)_{\text{COM}} = 0.856$$

$$P(s)_{\text{COUNT}} = 0.988 = 1 - q_{\text{COUNT}}$$

$$q_{\text{COUNT}} = 0.012$$

$$P(s)_{\text{SHIFT}} = 0.953 = 1 - q_{\text{SHIFT}}$$

$$q_{\text{SHIFT}} = 0.047$$

$$P(s)_{\text{subsystem}} = 0.856 (1 - 5.64 \times 10^{-4}) \approx 0.856$$

For 3 years considering four quadrants:

$$P(s) = 1 - q_{ss}^{-4}$$

$$q_{ss} = 1 - P(s)_{\text{subsystem 3 years}}$$

$$P(s) = 1 - (0.021) = 0.979$$

### Single-Mode FSK System

The reliability of the single-mode FSK system, like the dual-mode system, was calculated on the premise that no redundancy is used. Since there is only one mode of operation, the probability of success is determined by total part count only.

Parts breakdown:

|                                   | Total Parts<br>Count | Failure Rate,<br>(percent/<br>1000 hours) |
|-----------------------------------|----------------------|---|
| Carbon composition resistors      | 323                  | 0.323                                     |
| Carbon film resistors             | 31                   | 0.496                                     |
| Glass, mica, and paper capacitors | 76                   | 0.076                                     |
| Tantalum capacitors               | 15                   | 0.060                                     |
| Inductors                         | 12                   | 0.024                                     |

|                    | <u>Total Parts<br/>Count</u> | <u>Failure Rate,<br/>(percent/<br/>1000 hours)</u> |
|--------------------|------------------------------|--|
| Diodes             |                              |  |
| Switching          | 414                          | 0.828  |
| Zener              | 2                            | 0.030  |
| Transistors        | 64                           | <u>0.320</u>                                       |
| Total failure rate |                              | 2.157  |

Total system parts = 937

MTBF = 46,300 hours = 5.28 years

For 1 year considering one quadrant

$$P(s) = 0.83$$

For 3 years considering four quadrants

$$P(s) = 1 - q^4 = 1 - 0.04 = 0.96$$

#### Single-Mode Count System

The calculated reliability of the single-mode count system, like the preceding two types of systems considered, is based on the premise that no redundancy is used.

Parts breakdown and failure rates:

|                                   | <u>Total Parts<br/>Count</u> | <u>Failure Rate,<br/>(percent/<br/>1000 hours)</u> |
|-----------------------------------|------------------------------|--|
| Carbon composition resistors      | 286                          | 0.286  |
| Carbon film resistors             | 14                           | 0.224  |
| Glass, mica, and paper capacitors | 54                           | 0.054  |
| Tantalum capacitors               | 8                            | 0.032  |
| Inductors                         | 8                            | 0.016  |
| Diodes                            |                              |  |
| Switching                         | 391                          | 0.782  |
| Zener                             | 2                            | 0.030  |
| Transistors                       | 48                           | <u>0.240</u>                                       |
| Total failure rate                |                              | 1.664  |

Total system parts = 811

MTBF = 60,000 hours = 6.85 years

For 1 year considering one quadrant

$$P(s) = 0.864$$

For 3 years considering four quadrants

$$P(s) = 1 - 0.0127 = 0.983$$

It should be pointed out, at this time, that the parts counts given above are approximations, since the circuits have not yet been designed. The three systems are implemented in what seems to be a straightforward manner, but during the circuit development period it is possible that simplifications will be made. The reliability then, which is a function of parts count and part types, is also an approximation.

#### Comparison of Dual- and Single-Mode Systems

The dual-mode system contains more parts than either the count or shift single-mode systems. Its reliability is better than the single-mode FSK system, and not as good as the single-mode count system.

As stated earlier, redundancy was not considered in making the reliability calculations. It is believed that, while redundancy will increase the reliability of the single-mode systems, it will have a greater effect on the dual-mode system than on the other two.

As can be seen from the estimate made for the dual-mode system, the reliability is governed almost entirely by circuits that are common to and necessary for the operation of both modes. Increasing the reliability of the common circuitry will increase the reliability of the whole system by nearly the same amount, and possibly by adding a small number of parts the amount of common circuitry could be considerably reduced.

Since the operation of the single-mode systems requires that all circuitry operate, it may be considerably more difficult to improve their reliability significantly, or at least by as much as the dual-mode system could be improved.

Power dissipation would be nearly the same for the three different systems, and a comparison of part counts for the three subsystems shows differences on the order of 10 to 30 percent. It should be recognized, however, that the numbers which show the differences are only approximations.

Ultimately the decision as to the type of command decoder to be used rests with NASA. NASA's present inclination seems to be toward an FSK type of system. For this reason and also for reasons based on the preceding



discussion, it is recommended that the dual-mode system be used. It provides the simplicity of a count-type system combined with the capability of an FSK-type system which conforms to NASA's present desires. It gives reliability which is superior to a single-mode FSK system and which could possibly be better than a single-mode count-type system.

## APPENDIX B. JET CONTROL ELECTRONICS SUBSYSTEM DESIGN REVIEW

The functions of the jet control electronics (JCE) subsystem is either to generate a desired jet pulse envelope by sequential timing or permit a real-time jet pulse envelope to control the solenoid coil amplifiers of the two radial and two axial jets. The capabilities of the JCE and its general operation are described below.

With the antenna beam centered on the earth, a jet can be pulsed through 45 spacecraft degrees such that the average thrust direction of the radial jets will be +90 or -90 degrees from the spacecraft-earth line and an axial jet in phase and 180 degrees out of phase with the spacecraft-earth line. Also, both axial or radial jets can be pulsed with pulse envelopes 180 degrees out of phase with the result, for example, that radial jet No. 1 will be pulsed at +90 degrees and after one-half revolution, radial jet No. 2 will be pulsed at +90 degrees.

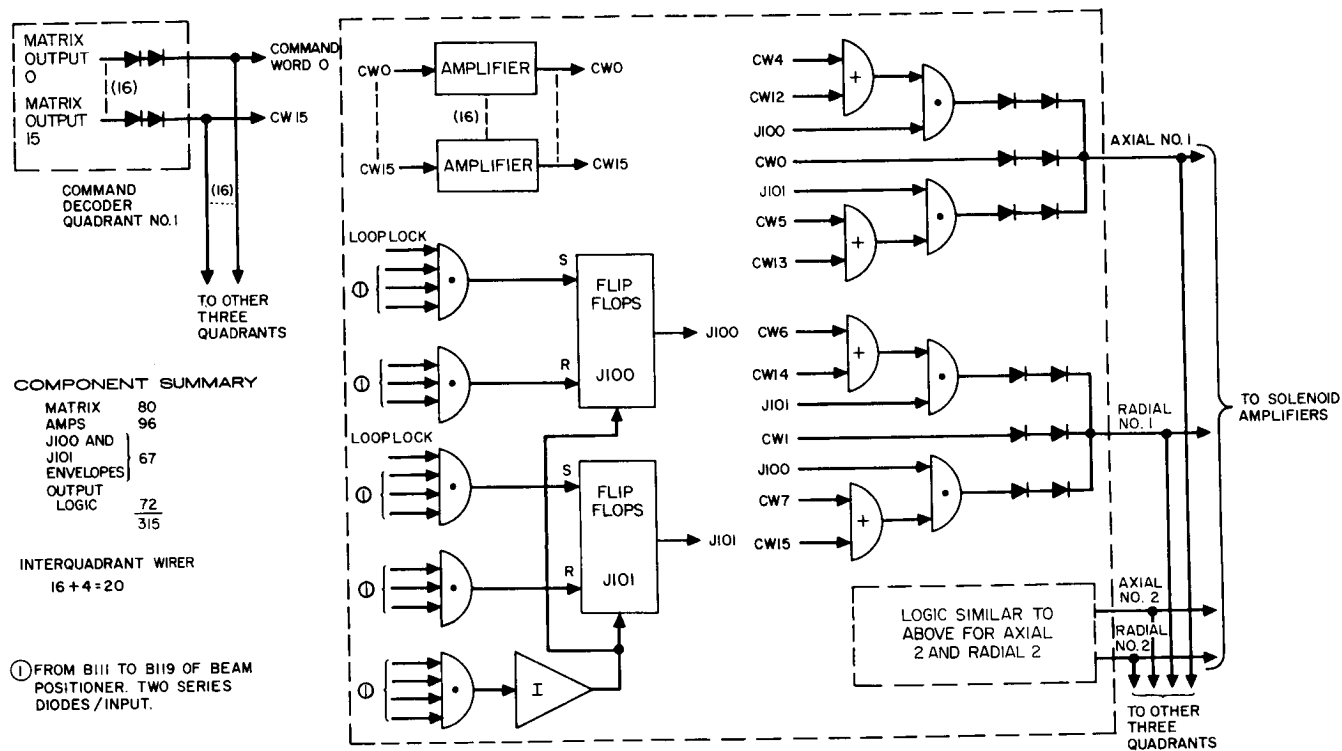
By rotating the antenna beam  $\theta$  degrees in the direction of the spin sense, the radial and axial jets can be sequentially pulsed +90 degrees +  $\theta$  degrees or -90 degrees +  $\theta$  degrees and  $\theta$  degrees or 180 degrees +  $\theta$  degrees, respectively, with respect to the spacecraft-earth line.

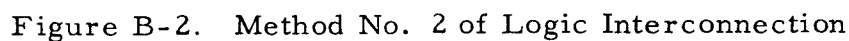
A backup mode, which is independent of the timing circuitry, is provided to pulse any jet at any angle relative to the sun or earth lines by real-time execution.

The basic timing for the pulsing circuitry is a counter which is earth-oriented; that is, a reference point on the periphery of the spacecraft will be coincident with the antenna beam center when the counter's content is zero. Therefore, if a reference jet is displaced  $\phi$  degrees from the reference point, it remains to detect a count of  $67.5 + \phi/360 \times 512$  to initiate the +90 degrees jet pulse. Detection of a count corresponding to 45 degrees later terminates the jet pulse. The jet pulse for a -90 degree correction is similarly obtained.

Sixteen command words are necessary to control the JCE. The command words and the functions of each are tabulated in Table B-1.

Three methods of interconnecting the JCE and the command decoder were considered. Figures B-1, B-2, and B-3 are the block diagrams of each. Their operations are, respectively:





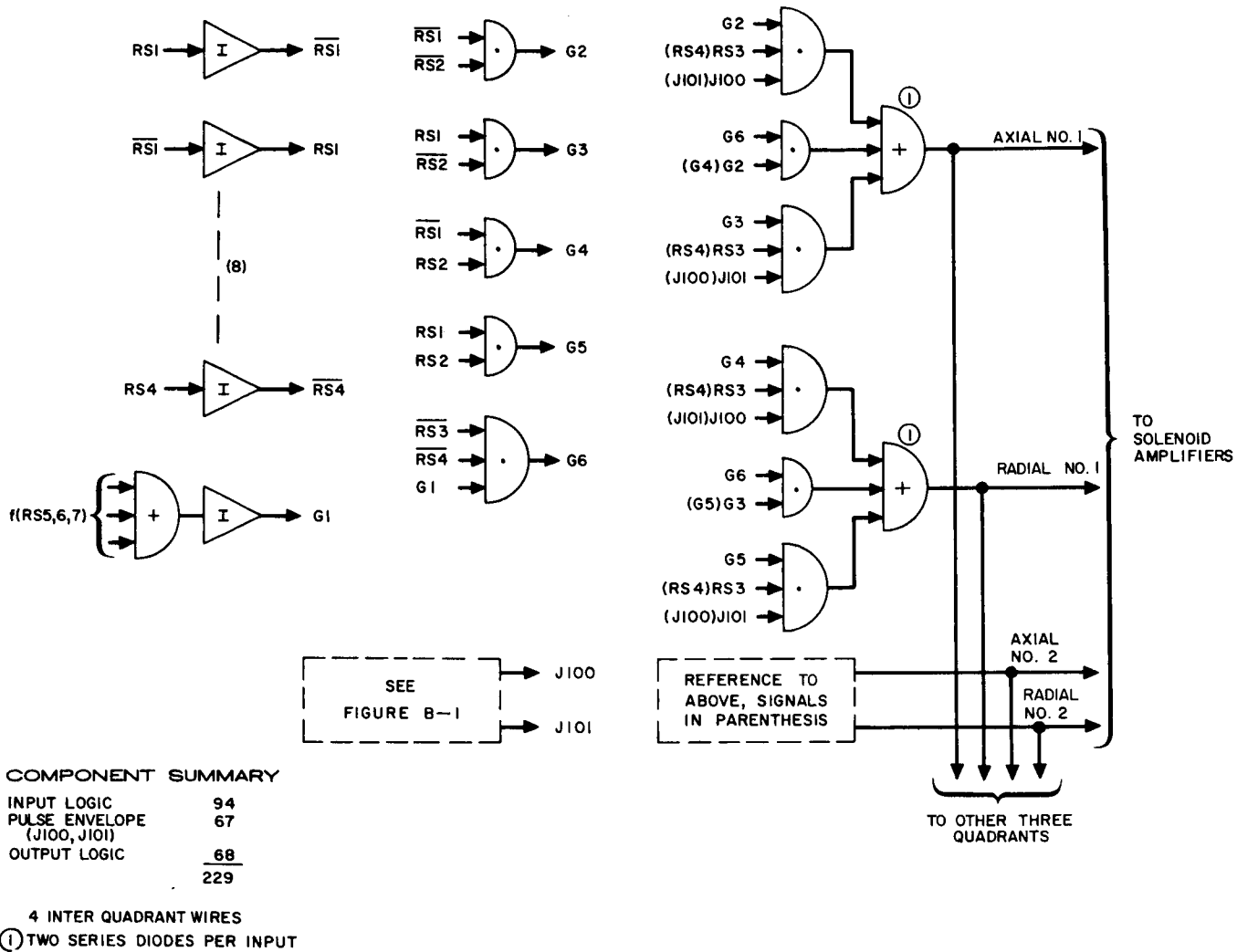


Figure B-3. Method No. 3 of Logic Interconnection

TABLE B-1. JET CONTROL ELECTRONICS COMMANDS

| Command Register Bits                       |     |     |     |     |     |     | Functions    |                      |
|---|-----|-----|-----|-----|-----|-----|--------------|----------------------|
| RS7   | RS6 | RS5 | RS4 | RS3 | RS2 | RS1 |              |                      |
| Function of these three bits to operate JCE |     |     | 0   | 0   | 0   | 0   | Axial No. 1  | Backup               |
|   |     |     | 0   | 0   | 0   | 1   | Radial No. 1 |                      |
|   |     |     | 0   | 0   | 1   | 0   | Axial No. 2  |                      |
|   |     |     | 0   | 0   | 1   | 1   | Radial No. 2 |                      |
|   |     |     | 0   | 1   | 0   | 0   | +90 axial    | Jet No. 1            |
|   |     |     | 0   | 1   | 0   | 1   | -90 axial    |                      |
|   |     |     | 0   | 1   | 1   | 0   | +90 radial   |                      |
|   |     |     | 0   | 1   | 1   | 1   | -90 radial   |                      |
|   |     |     | 1   | 0   | 0   | 0   | +90 axial    | Jet No. 2            |
|   |     |     | 1   | 0   | 0   | 1   | -90 axial    |                      |
|   |     |     | 1   | 0   | 1   | 0   | +90 radial   |                      |
|   |     |     | 1   | 0   | 1   | 1   | -90 radial   |                      |
|   |     |     | 1   | 1   | 0   | 0   | +90 axial    | Jets No. 1 and No. 2 |
|   |     |     | 1   | 1   | 0   | 1   | -90 axial    |                      |
|   |     |     | 1   | 1   | 1   | 0   | +90 radial   |                      |
|   |     |     | 1   | 1   | 1   | 1   | -90 radial   |                      |

- 1) Take the 16 commands from each command decoder matrix and "OR" their respective commands for distribution to the four jet control electronics subsystems.
- 2) Take the individual command bits from each command decoder and "OR" their respective bits for distribution to the JCE subsystems.
- 3) Drive a quadrant JCE with only its respective quadrant command decoder.

Using method 1, 20 interquadrant wires and 315 components would be necessary for implementation per quadrant.

Using method 2, 9 interquadrant wires and 259 components per quadrant are necessary.

Using method 3, 4 interquadrant wires and 229 components per quadrant are necessary.

A summary is given in Table B-2 along with other pertinent information.

TABLE B-2. SUBSYSTEM DESIGN SUMMARY

| Method | Inter-<br>Quadrant<br>Wires<br>per Quadrant | Components<br>per Quadrant | Combinations<br>of JCE-DC | Effectiveness<br>of Control |
|--------|---|----------------------------|---------------------------|-----------------------------|
| 1      | 20  | 315                        | 16                        | No<br>differences           |
| 2      | 9   | 259                        | 16                        |                             |
| 3      | 4   | 229                        | 4                         |                             |

A version of method No. 2, shown in Figure B-4, was picked. The difference is that the four combinations of RS1 and RS2, RS3, RS4, and the f(RS5, RS6, RS7) from each command decoder are OR'ed together and then distributed to the four quadrant JCE. The necessary components and wires per quadrant are 229 and 11, respectively.

It is necessary to detect the binary number corresponding to  $67.5 + \phi$  degrees to set flip-flop J100; 45 degrees later, J100 is reset; 180 degrees after J100 is set, flip-flop J101 is set, and 45 degrees later, J101 is reset. These operations are shown in Figure B-5.

Since, after setting J100, all other operations are a multiple of 45 degrees later, only B119, B118 and B117 need control the R&S inputs.

The inputs to the clock inverter (J526) are those necessary to detect a fraction of 45 degrees or  $\lambda$  shown in Figure B-5. The angle  $\lambda = \phi - 22.5$  degrees.

The signals J306P and F700P (loop lock) are also used to control the "S" input to J100 and J101. This is necessary to ensure that a jet pulse envelope is generated only when the JCE has been selected and the envelope timing signals are correct.

#### JET PULSE ERROR

As long as the frequency lock loop is in "LOCK" and the voltage-controlled oscillator is not increasing or decreasing in frequency during a revolution, no appreciable error in position or duration of the jet pulse control signal will result.

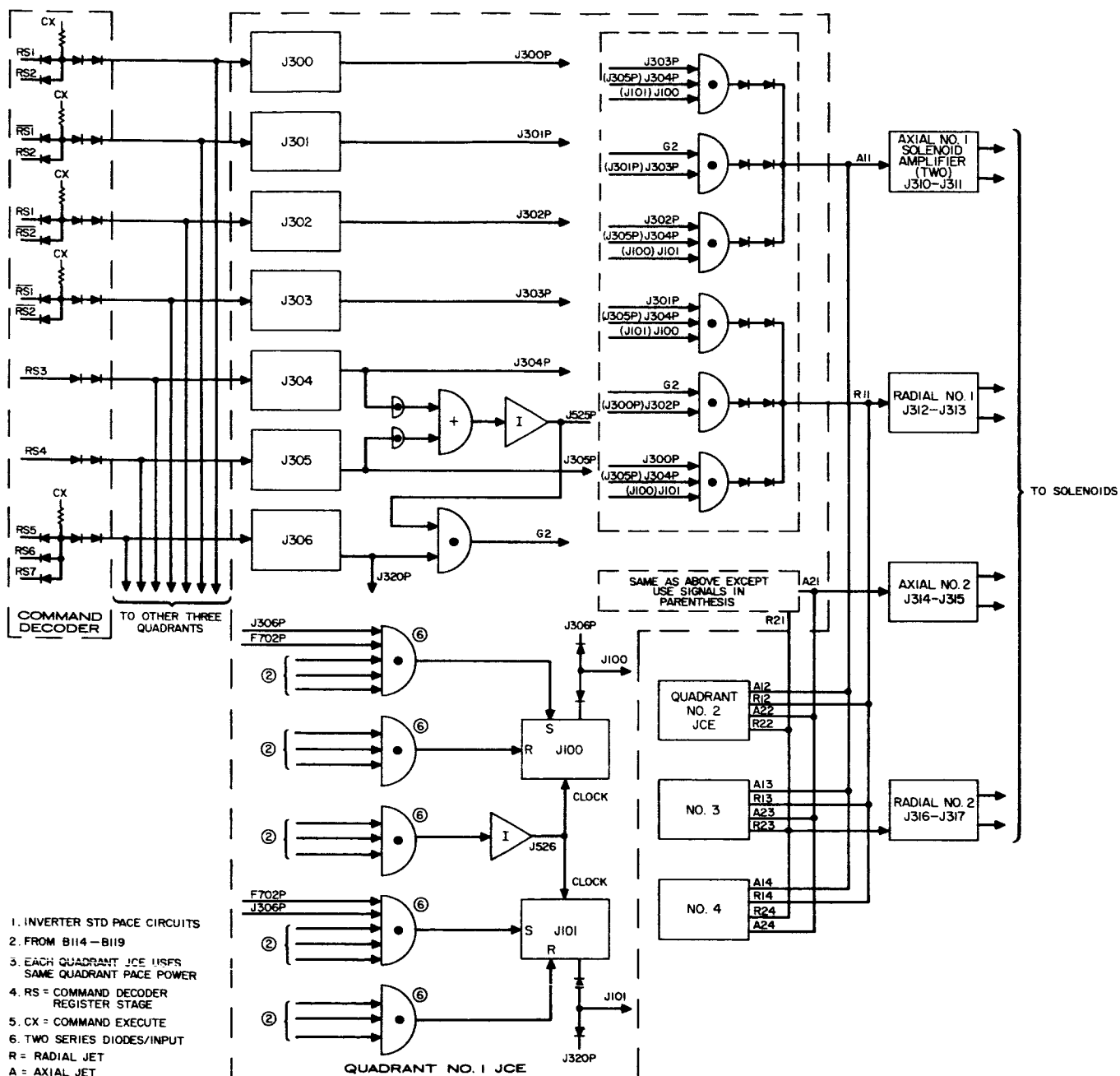


Figure B-4. Jet Control Electronics Final Diagram



## POWER

Standby - - - - 0.250 watts      Per JCE  
 Peak - - - - 40.000 watts  
 Average - - - - 3.50 watts      For 45 degree pulses

## PRELIMINARY COIL DATA

Valve closed inductance = 0.44 h  
 Valve open inductance = 1.11 h  
 Coil Resistance =  $39 \pm 2.5$   
 Maximum pull in current = 0.26 ampere  
 Maximum pull in time = 10 msec  
 Maximum holding current = 0.040 ampere

## LOGIC

Define flip-flops J100 and J101 with input equations as follows:

$$\left. \begin{aligned} J100S &= \frac{67.5 + \phi}{360} 512 \\ J100R &= \frac{112.5 + \phi}{360} 512 \\ J101S &= \frac{247.5 + \phi}{360} 512 \\ J101R &= \frac{297.5 + \phi}{360} 512 \end{aligned} \right\} \begin{array}{l} \text{Counts of} \\ \text{Beam} \\ \text{Positioner} \\ \text{Output} \\ \text{Counter} \end{array}$$

R = radial jet

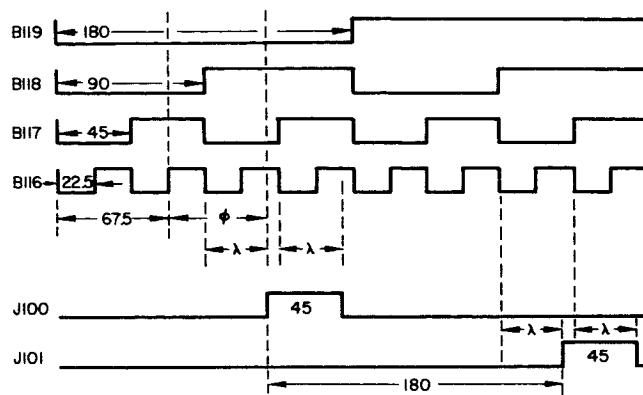
A = axial jet

Table B-3 is derived by inspection of Figure B-6.

TABLE B-3

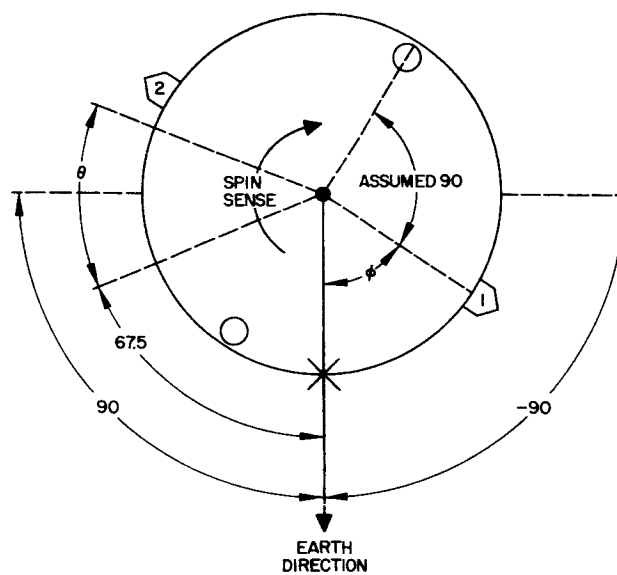
|     | (A1*)<br>R1 | (A2*)<br>R2 |
|-----|-------------|-------------|
| +90 | J100        | J101        |
| -90 | J101        | J100        |

\*After beam is repositioned.



DIMENSIONS IN DEGREES

Figure B-5. Inputs to Clock Inverter



ANGLES IN DEGREES

- X - REFERENCE POINT
- △ - RADIAL JET
- - AXIAL JET
- $\phi$  - ANGULAR DISPLACEMENT BETWEEN REFERENCE POINT AND RADIAL NO.1 JET
- $\theta$  - 90 deg PULSE ENVELOPE = 45 deg

Figure B-6. Top View of Spacecraft

Define command register bits

$$\begin{aligned}
 \overline{RS1} &= +90 \text{ degrees} & \overline{RS1} \cdot \overline{RS2} &= +90 (A) = J303P \\
 RS1 &= -90 \text{ degrees} & RS1 \cdot \overline{RS2} &= -90 (A) = J302P \\
 \overline{RS2} &= \text{axial} & \overline{RS1} \cdot RS2 &= +90 (R) = J301P \\
 RS2 &= \text{radial} & RS1 \cdot RS2 &= -90 (R) = J300P \\
 RS3 &= \text{No. 1 jet} & RS3 &= J304P \\
 RS4 &= \text{No. 2 jet} & RS4 &= J305P \\
 & & f(RS5, RS6, RS7) &= J306P \text{ (jet control operation)}
 \end{aligned}$$

## LOGIC EQUATIONS

Define:

$$\begin{aligned}
 G2 &= (\overline{J304P} + \overline{J305P}) \cdot J306P \\
 G3 &= J306P \cdot J100 \\
 G4 &= J306P \cdot J101 \\
 G2 \text{ is } &[ f(RS5, RS6, RS7) ] \cdot \overline{RS3} \cdot \overline{RS4}
 \end{aligned}$$

From Table B-3 the equations can be written by inspection.

$$\begin{aligned}
 \text{Radial No. 1} &= J301P \cdot J304P \cdot G3 + J300P \cdot J304P \cdot G4 \\
 \text{Radial No. 2} &= J301P \cdot J305P \cdot G4 + J300P \cdot J305P \cdot G3 \\
 \text{Axial No. 1} &= J303P \cdot J304P \cdot G3 + J302P \cdot J304P \cdot G4 \\
 \text{Axial No. 2} &= J303P \cdot J304P \cdot G4 + J302P \cdot J304P \cdot G3
 \end{aligned}$$

## For Backup Mode

$$\begin{aligned}
 \text{Radial No. 1} &= [ f(RS5, RS6, RS7) ] \cdot \overline{RS4} \cdot \overline{RS3} \cdot \overline{RS2} \cdot RS1 \\
 &= G2 \cdot J302P \\
 \text{Radial No. 2} &= [ f(RS5, RS6, RS7) ] \cdot \overline{RS4} \cdot \overline{RS3} \cdot RS2 \cdot RS1 \\
 &= G2 \cdot J300P \\
 \text{Axial No. 1} &= G2 \cdot J303P \\
 \text{Axial No. 2} &= G2 \cdot J301P
 \end{aligned}$$

These terms are OR'ed to those immediately above.

### BUFFER AMPLIFIER TYPE I (Figure B-7)

This circuit is of the emitter follower configuration which serves to transfer the low impedance level of the output circuits to a high impedance level for coupling to the command bits "OR" gate. Diode CR1 prevents any negative current flow from the three unused quadrants to the quadrant in use.

Since a collector to base short of Q1 would ground the command bits "OR" gate, CR2 is inserted to realize a circuit which requires two semiconductor failures in a quadrant to affect another quadrant.

### BUFFER AMPLIFIER TYPE II (Figure B-8)

This circuit is a double inverter with partial component redundancy. Partial component redundancy was necessary to realize a circuit which requires the failure to two components to effect a constant high-voltage output or a constant low-impedance input.

To result in a constant high-voltage output, Q2 and Q3 must open or Q1 and Q4 must short or R5 and R7 must short. For a constant low-impedance input to result, R1 and R2 must short.

### SOLENOID AMPLIFIER

The function of this amplifier is to provide a sufficiently high current pulse to saturate the solenoid and then supply a holding current for the remainder of the jet fire pulse. Another criterion is to design a circuit which necessitates the failure of two semiconductors to activate a solenoid without a command.

The output transistors, Q8 and Q9, are driven by two separate pre-amplifiers, whose transistors are either all on during command execution or all off. With separate preamplifiers and output transistors, one transistor from Q1, Q3, Q5, Q8 and one from Q2, Q4, Q6, Q9 must short to result in activation of the solenoid without a control signal.

As mentioned, it is desired to supply first a pull-in current and then the holding current after the electronic switch is closed. The circuit of Figure B-9 accomplishes this if the capacitance of C is large enough to appear as essentially a short during the high-current duration. When C charges to a voltage determined by  $R_A$ ,  $R_B$ , and R coil, only the relatively small holding current is allowed to flow. However, to effect an essential short by C during the high-current duration, C would be prohibitively large. Transistor Q7 performs as a capacitor multiplier in that it limits the capacitor charging current by a factor of  $1/\beta_{Q7}$ , and thus permits the use of a much smaller component.

The coil current-time envelope is as shown in Figure B-10. It should be recognized from the current plot that if Q8 and Q9 either shorts or saturates due to a failure in the preamplifiers, the steady-state current drain from the power system will be  $I_{LH}$ .

When the jet pulse is terminated, C1 and C2 discharge through R16, CR2, and CR3.

Diodes CR4 through CR7 are used to decrease the delay time of the coil current when the jet pulse is terminated.

A component summary is given in Table B-4.

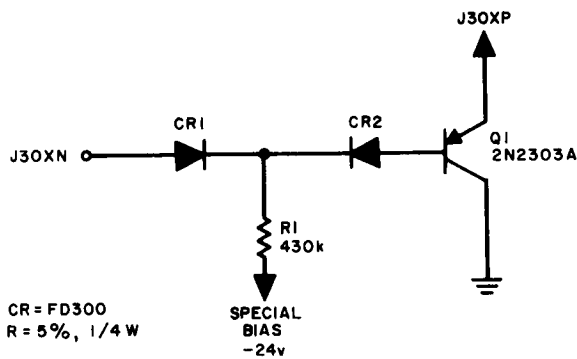


Figure B-7. Buffer Amplifier Type I

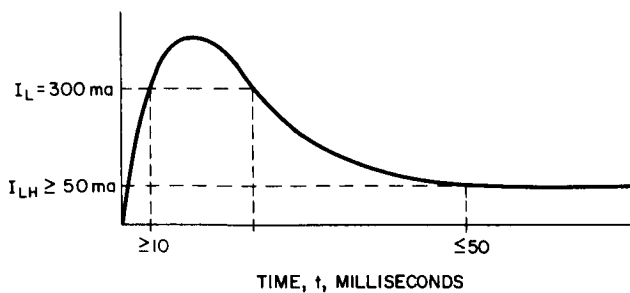


Figure B-10. Coil Current Pulse Envelope

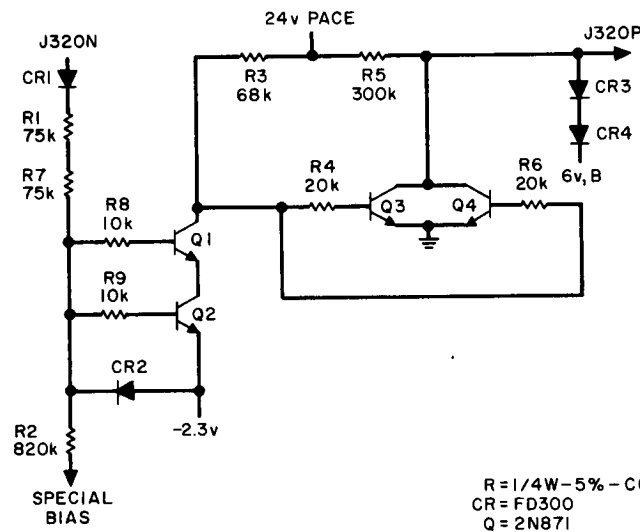


Figure B-8. Buffer Amplifier Type II

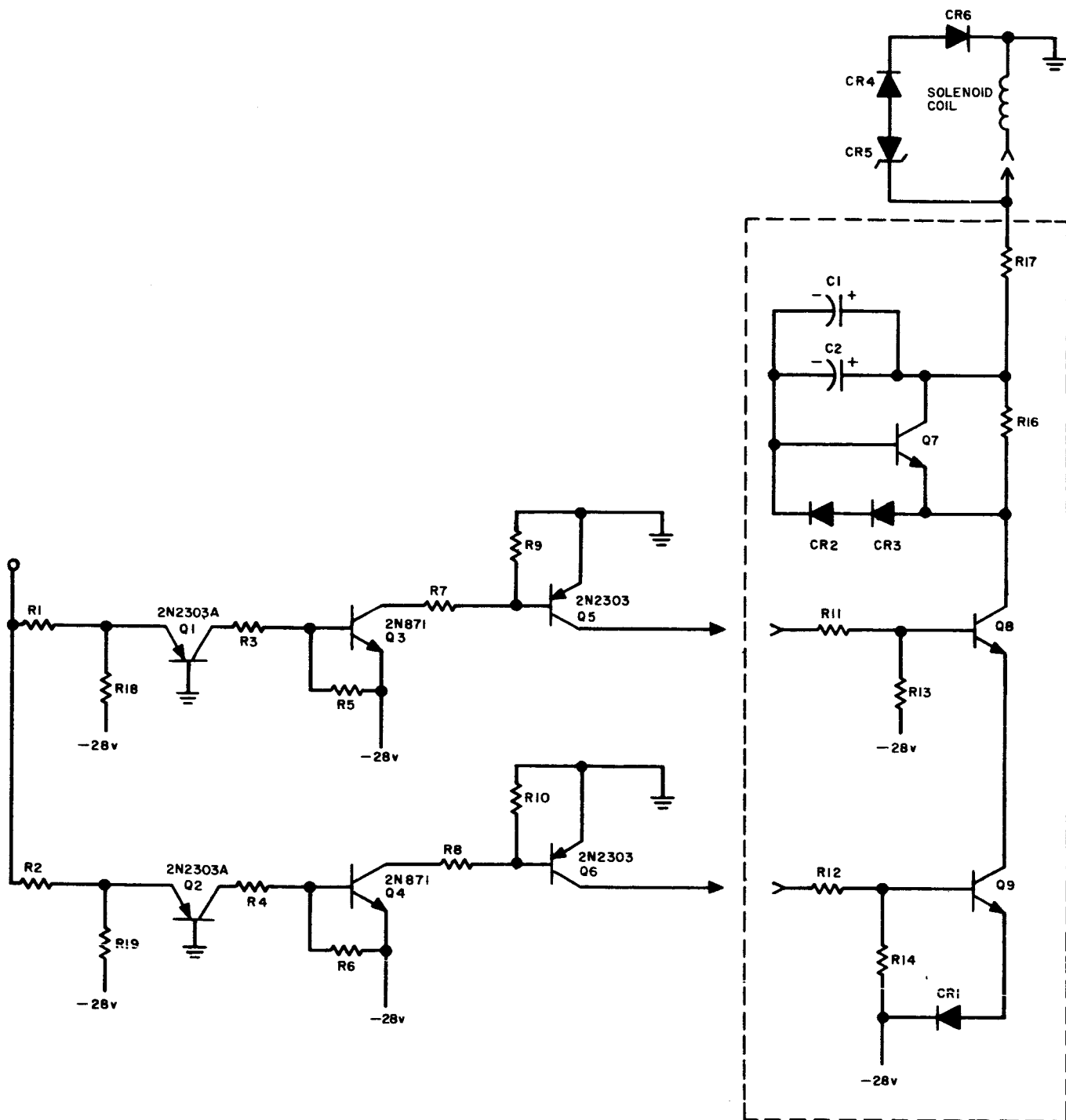


Figure B-9. Solenoid Amplifier

TABLE B-4. COMPONENT SUMMARY

| Circuit                  | Circuits per Quadrant | Transistors |       |         |        |         | Diodes |        |        | Resistors                        |                                  |                      |                     | Capacitors             |    | Components per Circuit | Components per Quadrant |
|--------------------------|-----------------------|-------------|-------|---------|--------|---------|--------|--------|--------|----------------------------------|----------------------------------|----------------------|---------------------|------------------------|----|------------------------|-------------------------|
|                          |                       | 2N2484      | 2N871 | 2N2193A | 2N2303 | 2N2303A | FD300  | 1N3189 | 1N2976 | Resistor<br>5% - $\frac{1}{4}$ W | Resistor<br>5% - $\frac{1}{8}$ W | Resistor<br>5% - 2 W | Glass<br>5% - 300 V | Tantalum<br>10% - 50 V |    |                        |                         |
| Command decoder gates    | 7                     |             |       |         |        |         | 4      | 28     |        | 1                                | 7                                |                      |                     |                        | 5  | 35                     |                         |
| Buffer amplifier Type I  | 6                     |             |       |         |        | 1       | 2      | 12     |        | 1                                | 6                                |                      |                     |                        | 4  | 24                     |                         |
| Buffer amplifier Type II | 1                     | 4           | 4     |         |        |         | 4      | 4      |        | 9                                | 9                                |                      |                     |                        | 17 | 17                     |                         |
| Special inverter         | 1                     | 1           |       |         |        |         | 5      | 5      |        | 4                                | 4                                | 1                    |                     |                        | 11 | 11                     |                         |
| Standard inverter        | 1                     | 1           |       |         |        |         | 2      | 2      |        | 3                                | 3                                |                      |                     |                        | 6  | 6                      |                         |
| Flip-flops               | 2                     | 2           | 4     |         |        |         | 8      | 16     |        | 4                                | 8                                |                      | 2                   | 4                      | 16 | 32                     |                         |
| Three-term logic gates   | 11                    |             |       |         |        |         | 3      | 33     |        | 1                                | 11                               |                      |                     |                        | 4  | 44                     |                         |
| Two-term logic gates     | 6                     |             |       |         |        |         | 2      | 12     |        | 1                                | 6                                |                      |                     |                        | 3  | 18                     |                         |
| Five-term logic gates    | 2                     |             |       |         |        |         | 5      | 10     |        | 1                                | 2                                |                      |                     |                        | 6  | 12                     |                         |
| Special bias             | 1                     | 2           | 2     |         |        |         | 1      | 1      |        | 3                                | 3                                |                      |                     |                        | 6  | 6                      |                         |
| Output "OR" gates        | 4                     |             |       |         |        |         | 6      | 24     |        |                                  |                                  |                      |                     |                        | 6  | 24                     |                         |
|                          |                       |             |       |         |        |         |        |        |        |                                  |                                  |                      |                     |                        |    |                        |                         |
| Solenoid amplifiers      | 4                     | 2           | 8     | 3       | 2      | 2       | 2      | 8      | 3      | 1                                | 14                               | 6                    | 24                  | 2                      | 8  | 37                     | 148                     |
|                          |                       |             |       |         |        |         |        |        |        |                                  |                                  |                      |                     |                        |    |                        |                         |
|                          |                       | 6           | 6     |         |        |         | 6      | 147    |        |                                  | 59                               | 1                    |                     | 4                      |    |                        | 229*                    |

\*Does not include solenoid amplifier.